PHASE 0/I FLIGHT SAFETY DATA PACKAGE FOR THE ALPHA MAGNETIC SPECTROMETER-02 (AMS-02)

Space and Life Sciences Directorate
Flight Projects Division

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National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston, Texas

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FOR THE

ALPHA MAGNETIC SPECTROMETER-02 (AMS-02)

NASA/JSC APPROVAL

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Flight Projects Division

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS

JANUARY 2001

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FOR THE

ALPHA MAGNETIC SPECTROMETER-02 (AMS-02)

Technical Work Plan Number: MHECSMBSX

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Contract NAS 9-19100

For

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS

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TABLE OF CONTENTS

Table of Contents, Continued

TABLE OF CONTENTS (Continued)

FIGURES

FIGURES (Continue)

FIGURES (Continue)

FIGURES (Continue)

FIGURES (Continue)

APPENDICES

ACRONYMS AND ABBREVIATIONS

ACC	Anti-Coincidence Counter
ACOP	AMS Crew Operations Post
AMS-02	Alpha Magnetic Spectrometer-02
APCU	Assembly Power Converter Unit
BCS	Berthing Cues System
BD	Burst Disk
μCi	micro Currie
CF_4	Freon
CO_2	Carbon Dioxide
COTS	Commercial Off the Shelf
CU	Copper
DC	Direct Current
DDRS	Digital Data Recording System
ECAL	Electromagnetic Calorimeter
EMI	Electromagnetic Interference
EMU	Extravehicular Mobility Unit
ETH	Eidgenossische Technische Hochschule
EVA	Extravehicular Activity
Fe^{55}	Iron 55
FRGF	Flight Releasable Grapple Fixture
G	Gravity (also g)
GeV	Giga Electron Volts

Grms	Gravity Root Mean Square
HC	Honeycomb
HDBK	Handbook
HP	High Pressure
HRDL	High Rate Date Link
Hz	Hertz
ICD	Interface Control Document
ID	Inner Diameter
ISIS	International Subrack Interface Standard
ISS	International Space Station
IVA	Intravehicular Activity
JSC	Johnson Space Center
K	Kelvin
KeV	Kilo Electron Volts
L	Liter (also l)
LCD	Liquid Crystal Display
LMSO	Lockheed Martin Space Operations
LP	Low Pressure
μm	micrometer
m	meter
mm	millimeter
MCC	Monitoring and Control Computer
MDP	Maximum Dynamic Pressure

MLI	Multilayer Insulation
MOD	Meteoroid and Orbital Debris
MSFC	Marshall Space Flight Center
MUA	Materials Usage Agreement
NASA	National Aeronautics and Space Administration
NH_3	Ammonia
NHB	NASA Handbook
NSTS	National Space Transportation System
O	Orifice
OIU	Orbiter Interface Unit
PAS	Payload Attach System
PC	Printed Circuit
PCI	Peripheral Connect Interface
PCS	Portable Computer System
PDIP	Payload Data Interface Panel
PEDS	Passive Electrical Disconnect System
PGSC	Payload and General Support Computer
PMT	Photo Multiplier Tube
PDB	Power Distribution Box
PVGF	Power Video Grapple Fixture
RF	Radio Frequency
RICH	Ring Imaging Cherenkov Counter
ROEU	Remotely Operated Electrical Umbilical

RPM	Revolutions Per Minute
SCL	Space Cryomagnetics Limited
SCSI	Small Computer Systems Interface
SFHe	Superfluid Helium (aka. He^{II})
SPEC	Specification
SRD	Synchrotron Radiation Detector
SRMS	Space Shuttle Remote Manipulator System
SSRMS	Space Station Remote Manipulator System
SSP	Space Shuttle Program
SSP	Standard Switch Panel
STD	Standard
STS	Space Transportation System
TAS	Tracker Alignment System
TBD	To Be Determined
TBS	To Be Supplied
Te	Tellurium
TeV	Tera Electron Volts
TMP	Thermal Mechanical Pump
TOF	Time Of Flight
UIP	Utility Interface Panel
UMA	Umbilical Mechanism Assembly
UPS	Unswitched Power Supply or Uninterruptible Power Supply
USS-02	Unique Support Structure-02
Xe	Xenon

APPLICABLE SAFETY DOCUMENTS

NSTS 1700.7B	Safety Policy and Requirements For Payloads Using the Space Transportation System
NSTS 1700.7B ISS Addendum	Safety Policy and Requirements For Payloads Using the International Space Station
NSTS/ISS 13830C	Payload Safety Review and Data Submittal Requirements For Payloads Using the Space Shuttle/International Space Station
NSTS/ISS 18798B	Interpretations of NSTS/ISS Payload Safety Requirements
JSC 26943	Guidelines for the Preparation of Payload Flight Safety Data Packages and Hazard Reports For Payloads Using the Space Shuttle

1.0 INTRODUCTION

This Phase 0/I Flight Safety Data Package for the Alpha Magnetic Spectrometer-02 (AMS-02) is submitted in response to the safety requirements of NSTS 1700.7B, "Safety Policy and Requirements For Payloads Using the Space Transportation System", and NSTS 1700.7B, ISS Addendum, ”Safety Policy and Requirements For Payloads Using the International Space Station. This safety package has been prepared in accordance with NSTS/ISS 13830C, "Payload Safety Review and Data Submittal Requirements For Payloads Using the Space Shuttle/International Space Station”. Also, JSC 26943, "Guidelines for the Preparation of Payload Flight Safety Data Packages and Hazard Reports For Payloads Using the Space Shuttle", was used as a guideline document.

2.0 SCOPE

This safety package contains the safety analysis performed for the AMS-02 flight hardware (See Figures 2.1 thru 2.4) and flight operations. The major subsystems of the AMS-02 covered by this safety analysis are the Cryogenic Superconducting Magnet (Cryomag), Unique Support Structure-02 (USS-02) (which includes the integral vacuum case for the cryomag), Synchrotron Radiation Detector (SRD), Transition Radiation Detector (TRD), Time-Of-Flight (TOF) Scintillator Assemblies, Ring Imaging Cherenkov Counter (RICH), Electromagnetic Calorimeter (ECAL), data and interface electronics, electrical cables, two Monitoring and Control Computers (MCCs), Power Distribution Box, AMS Crew Operations Post (ACOP), Thermal Control System (TCS), Meteoroid and Orbital Debris (MM&OD) shields and passive Payload Attach System (PAS).

The AMS-02 Silicon Tracker Assemblies, Tracker Alignment System (TAS), Anti-Coincidence Counter (ACC) and Digital Data Recording System (DDRS) are reflown/series elements from AMS-01. The reflown/ series safety assessment for these elements will be included in the AMS-02 Phase III Flight Safety Data Package.

The Space Shuttle Program (SSP) provided hardware that will be used with the AMS-02 includes: Two Orbiter Interface Units (OIU's), two PGSCs with expansion assemblies and power cables, Flight Releasable Grapple Fixture (FRGF) and Remotely Operated Electrical Umbilical (ROEU) or Passive Electrical Disconnect System (PEDS). The International Space Station (ISS) Program provided hardware that will be used with the AMS-02 includes: Two Assembly Power Converter Units (APCU's), 4 Panel-Unit (PU) International Subrack Interface Standard (ISIS) drawer in the ISS EXPRESS Rack, Power Video Grapple Fixture (PVGF), passive Umbilical Mechanism Assembly (UMA), Berthing Cues System (BCS). The AMS-02 payload will also require the use of the Shuttle Remote Manipulator System (SRMS), the Space Station Remote Manipulator System (SSRMS), the active PAS, and active UMA. The safety analyses for the SSP and ISS provided hardware are not part of this safety data package. However, the safety of the use and interfaces of the SSP and ISS provided hardware with the AMS-02 payload are part of this AMS-02 safety data package.

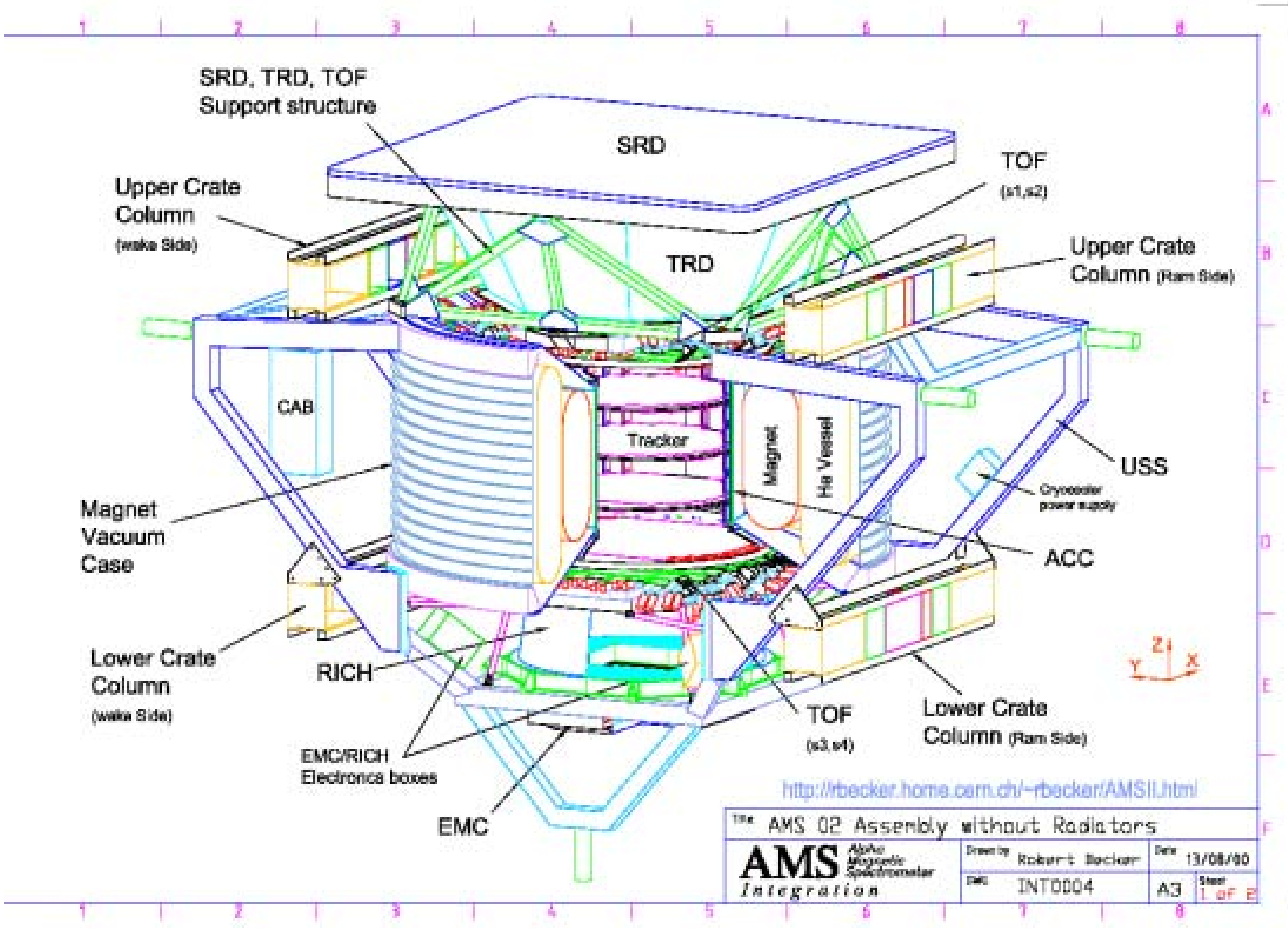


Figure 2.1 – AMS-02 with the USS-02 (Sheet 1 of 2)

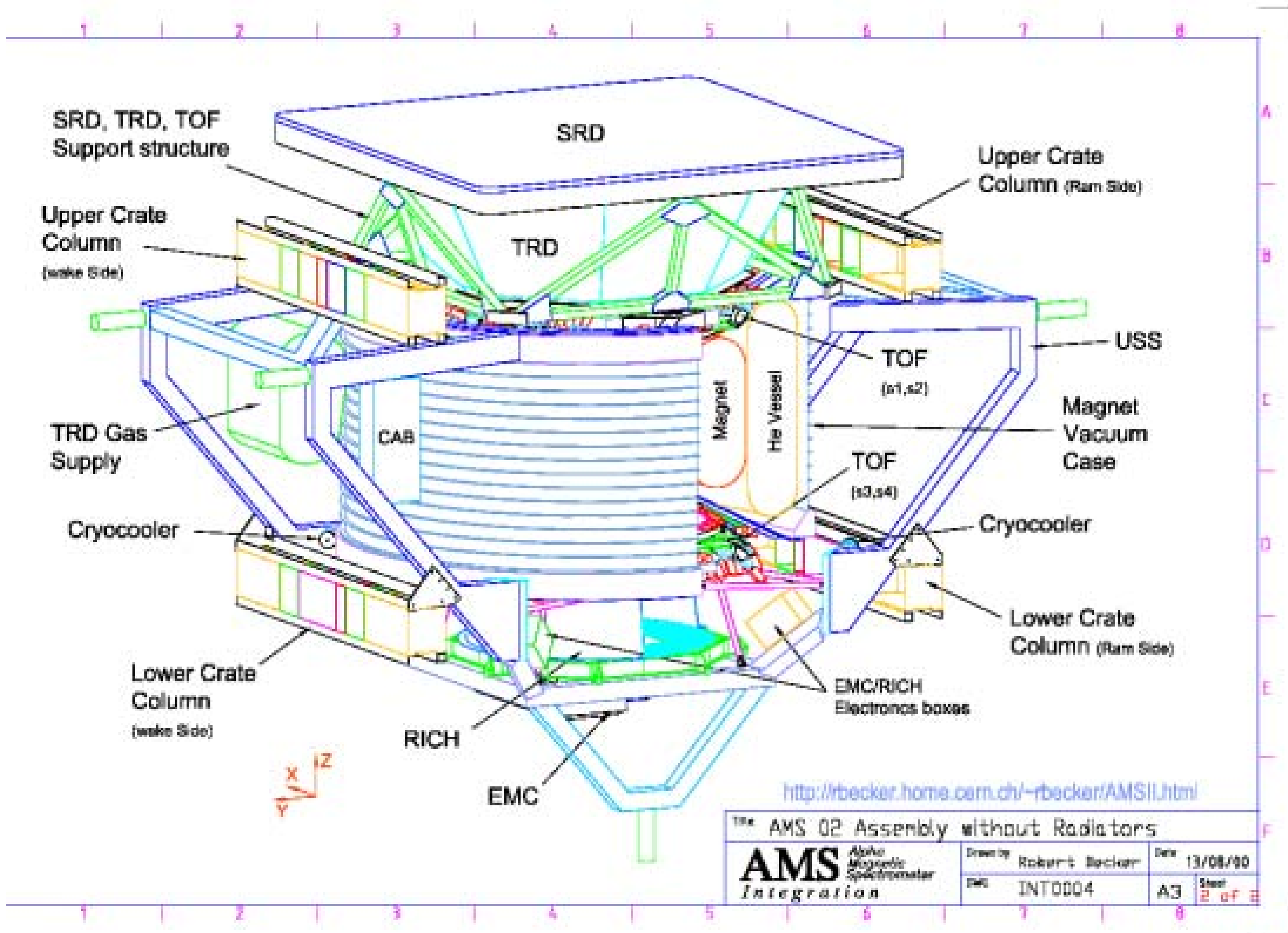


Figure 2.2 – AMS-02 with the USS-02 (Sheet 2 of 2)

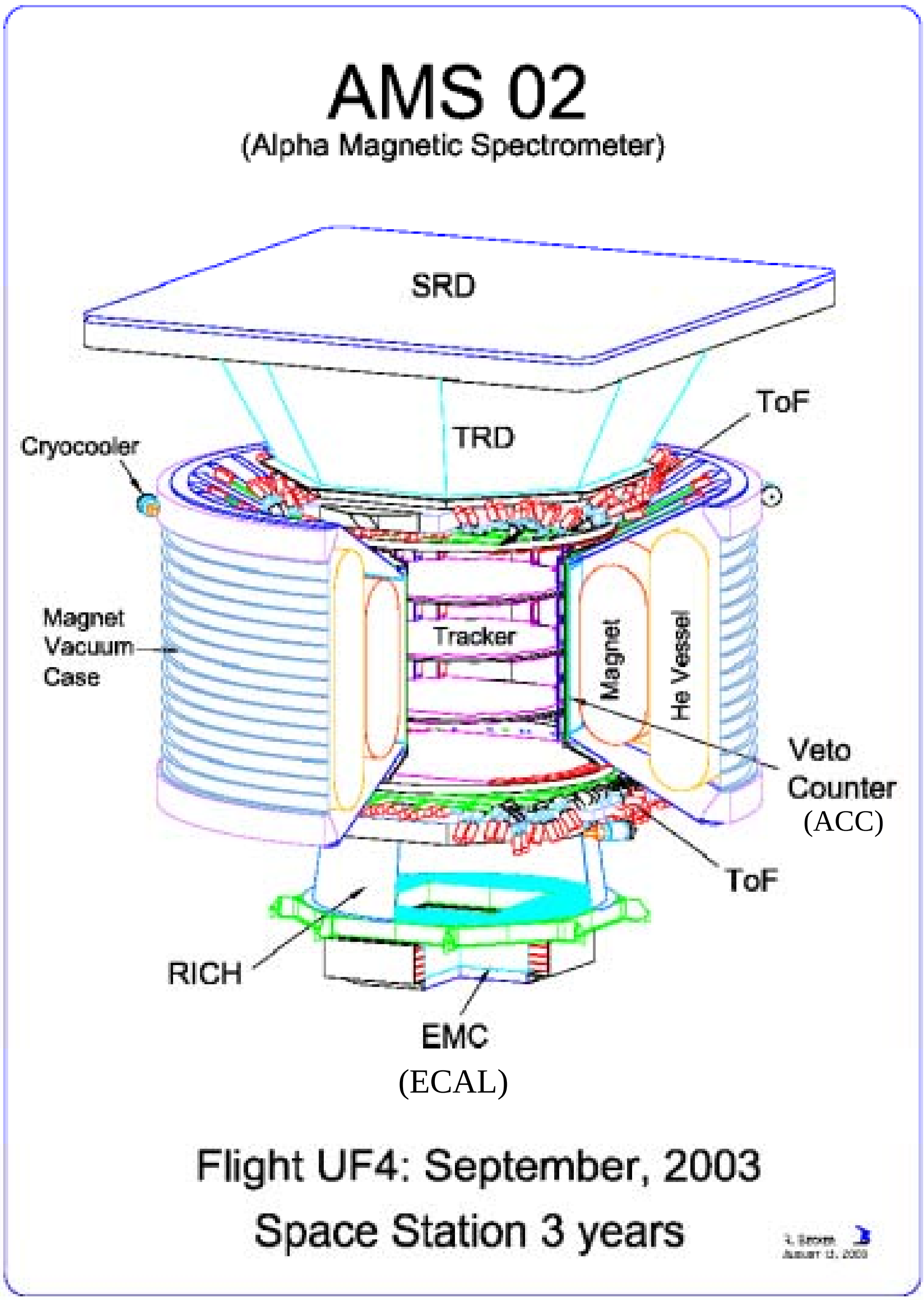


Figure 2.3 – AMS-02 without the USS-02 (Sheet 1 of 2)

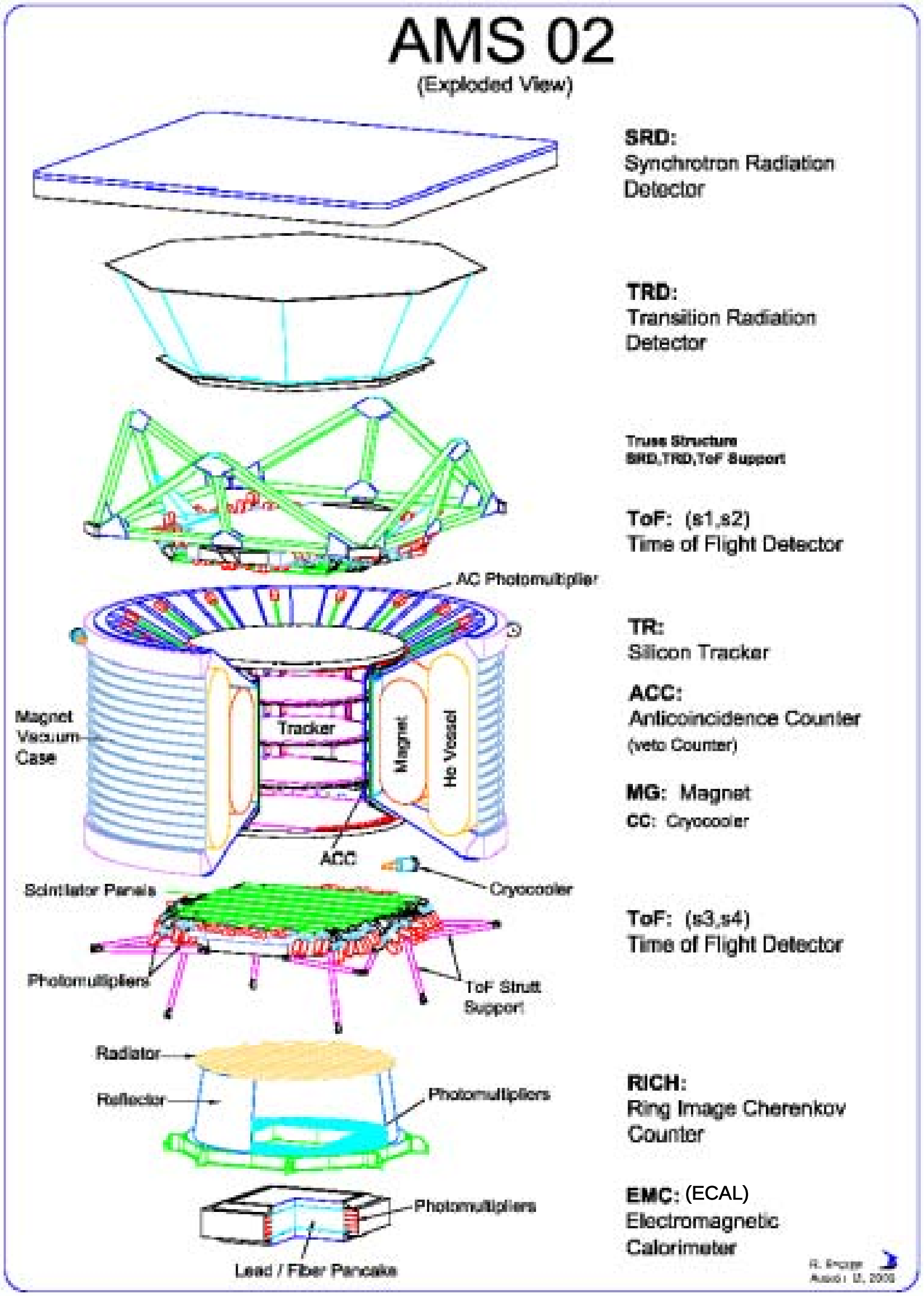


Figure 2.4 – AMS-02 without the USS-02 (Sheet 2 of 2)

3.0 PURPOSE

The purpose of this safety analysis is to identify potential flight hazards associated with the AMS-02, to evaluate their cause and impact on the Space Shuttle, ISS and flight crews, to define methods for eliminating or controlling the hazards, to verify the elimination or control methods, and to document the status of the verification methods. This safety package is intended to provide the information necessary for a Phase 0/I review of the AMS-02 by the Space Shuttle Payload Flight Safety Review Panel.

4.0 AMS-02 PAYLOAD OVERVIEW

The AMS-02 experiment is a state-of-the-art particle physics detector. The science objectives of the AMS-02 experiment are to search for antimatter (anti-helium and anti-carbon) in space, to search for dark matter (90% of the missing matter in the universe) and to study astrophysics (to understand Cosmic Ray propagation and confinement time in the Galaxy).

The AMS-02 Experiment will utilize a Cryogenic Superconducting Magnet (Cryomag) with planes of detectors on top, inside and below the magnet. Electrically charged particles which pass through the magnetic field will curve. Cosmic rays made of matter will curve one way, and those of antimatter will curve the opposite way. The positions of electrons released as the charged particles pass through the detectors will be electronically recorded. Physicists will be able to study the trajectory of curvature and determine the charge of the particles from the direction of curvature. They will also be able to determine the mass of the particles from the amount of curvature. They will then be able to tell whether it was matter or antimatter.

The Space Shuttle flight of AMS-01 was a precursor flight of some of the detectors with a permanent magnet in place of a cryomagnet. The purpose of the precursor flight was to verify operation of the AMS experiment, verify command/data communications, collect thermal data for the ISS flight, determine actual accelerations on some AMS internal instruments and establish experimental background data. The Space Shuttle flight of AMS-02 (See Figure 4.1) is a flight to install the AMS-02 on the external truss of the International Space Station (ISS) (See Figures 4.2 & 4.3), where it is scheduled to remain for at least three operational years of data collection.

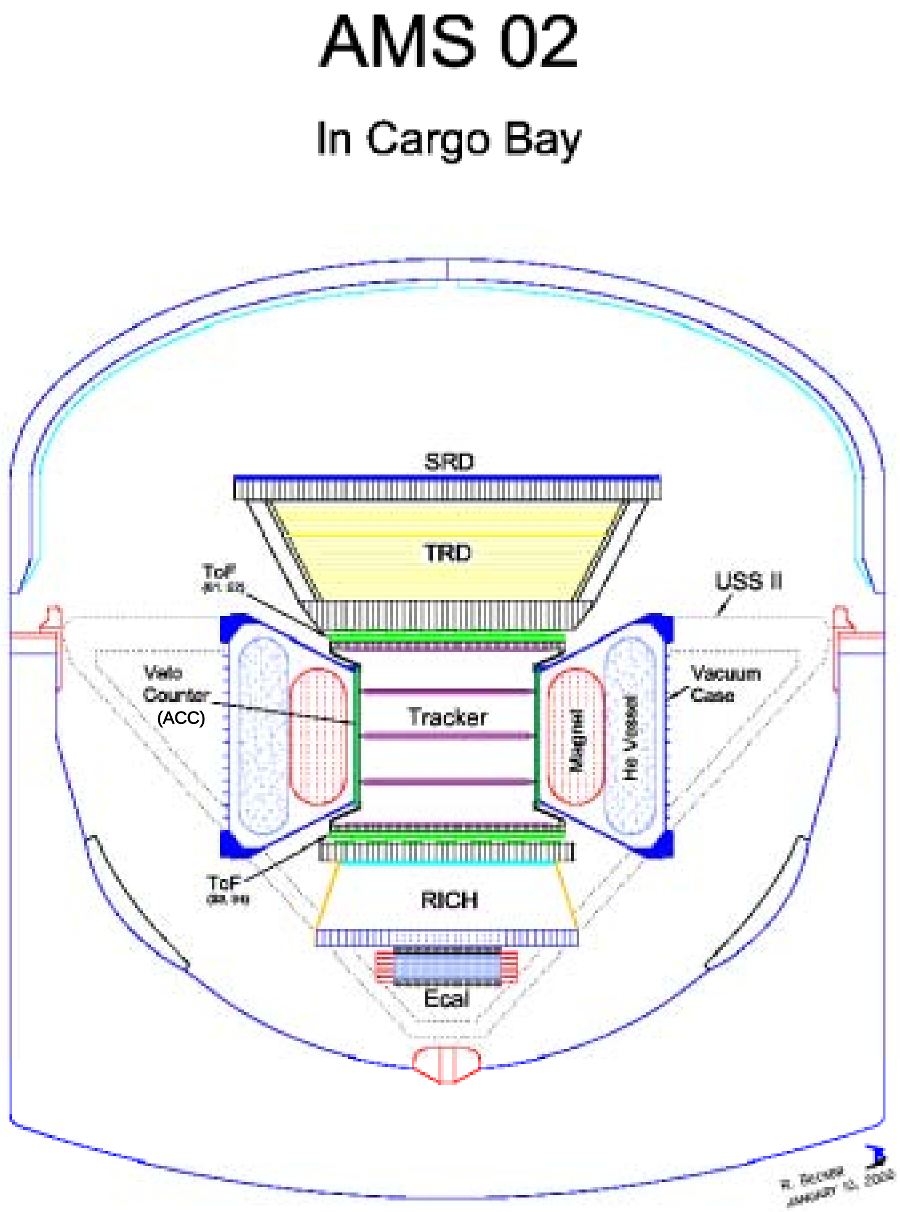


Figure 4.1 – AMS-02 in the Space Shuttle Orbiter Cargo Bay

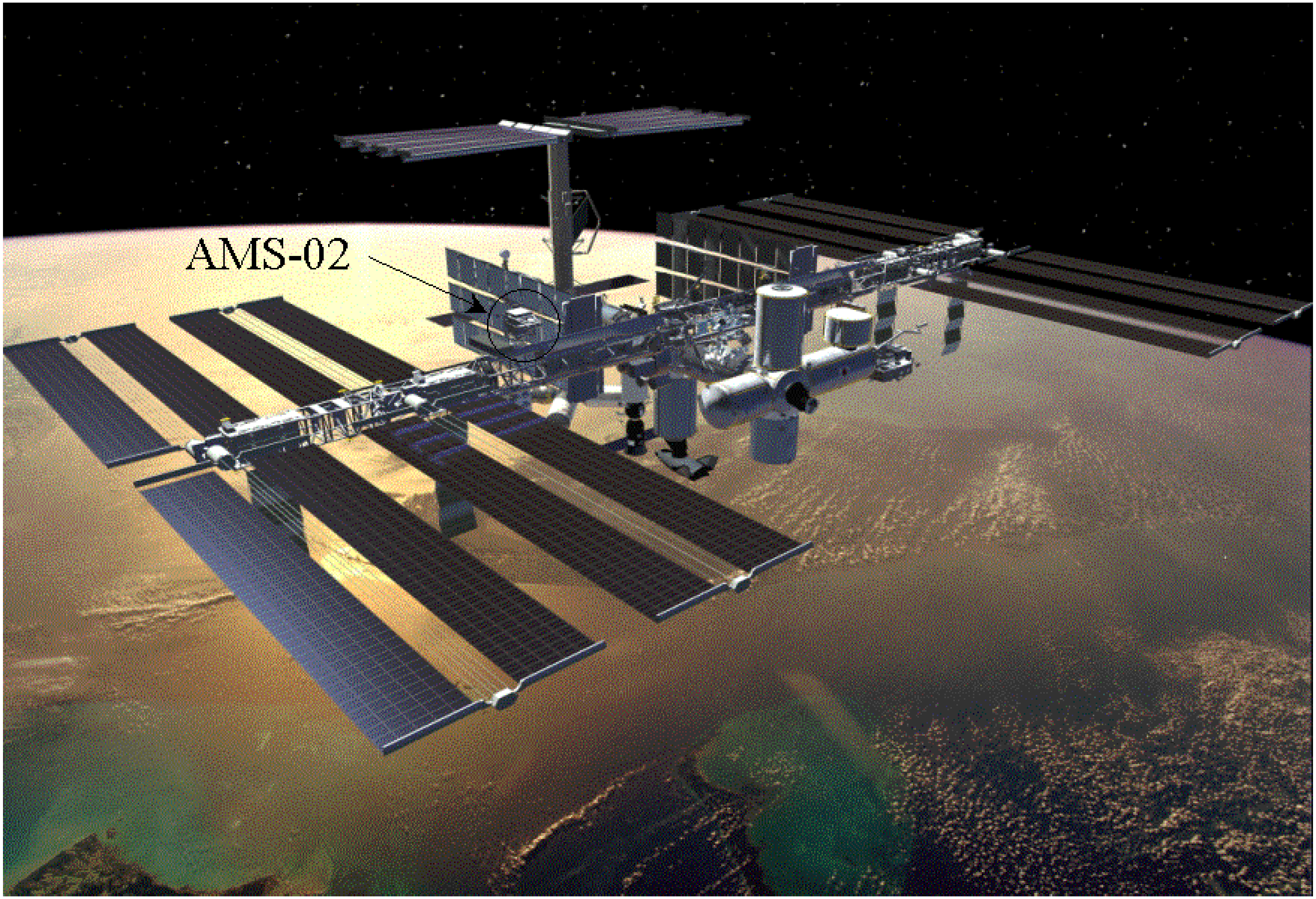


Figure 4.2 – AMS-02 on the ISS

Figure 4.3 – AMS-02 Payload Assembly on ISS S3 – Z Inboard PAS Site

5.0 AMS FLIGHT HARDWARE DESCRIPTION

5.1 CRYOGENIC SUPERCONDUCTING MAGNET (CRYOMAG)

5.1.1 Cryomag Description

Most of the cryomag and any cryomag related special test equipment (STE) will be developed and manufactured by Eidgenossische Technische Hochschule (ETH) in Zurich through a sub-contract. All of the design and analysis technical support is provided by ETH. The system will consist of a large superconducting magnet and a large Superfluid Helium (SFHe) dewar. ETH will also provide any Ground Handling Equipment (GHE) and any Ground Support Equipment (GSE) for cryogenic systems filling, servicing, monitoring, or control. This ETH provided hardware will be used in England, Zurich, KSC, and possibly JSC. The total magnet system including the vacuum case is currently estimated to weigh ~6508 lbs (2952 Kg).

The Lockheed Martin Advanced Technology Center (LMATC) of Palo Alto, California may also provide additional hardware and development support to ETH on a separate contract.

The cryocoolers for the Cryomagnet will most likely be built by Sunpower (Ohio) and will be certified for flight by the Cryo systems group at Goddard Space Flight Center (GSFC). There is no fluid path connecting the cryocoolers to the magnet cryosystem. All cooling is accomplished through conduction.

LMSO will provide analysis and design for the cryomag vacuum case hardware. The Vacuum Case (VC) (See Figures 5.1.1.1, 5.1.1.2 & 5.1.1.3) (weight currently estimated at 1457 lbs (661 Kg).) serves a dual purpose. It is a primary structural support and works in conjunction with the USS-02. In addition, it serves as a vacuum vessel for the cryosystem and magnet which is suspended inside the VC by sixteen support straps. The toroidal VC will be made of aluminum 2219 and 7050. The main structural components of the VC are: upper conical flange, lower conical flange, upper support ring, lower support ring, inner cylinder and outer cylinder. On the VC inner joints, the upper and lower flanges will be butt welded to the inner cylinder (See Figure 5.1.1.4). On the VC outer joints, the upper and lower flanges will be attached to the outer cylinder with 192 fasteners at each interface. Double O-rings will be used at each outer joint interface (See Figure 5.1.1.5). The O-ring material is currently Viton, but may be changed to Silicone, pending thermal analysis results. The O-rings are cord stock that has been joined at seams using the Parker hot vulcanizing process. There will be test ports between the O-rings to test them. There will be 41 ports in the vacuum case. Sixteen ports will be for the cryosystem support straps and 25 ports will be for plumbing lines, burst disks, electrical connections, etc. (See Figures 5.1.1.6 thru 5.1.1.9). Double O-rings will be used at each of the ports. All pump-out ports will also have redundant seals. The VC will also have 3 burst disks in series for emergency venting. Since the VC is considered to be an integral primary structure with the USS-02 for all structural design, analyses, testing, and safety assessments, a Structural Test Article (STA) of the vacuum case will be developed and used during much of the AMS-02 testing. The flight unit and STA unit will be identical.

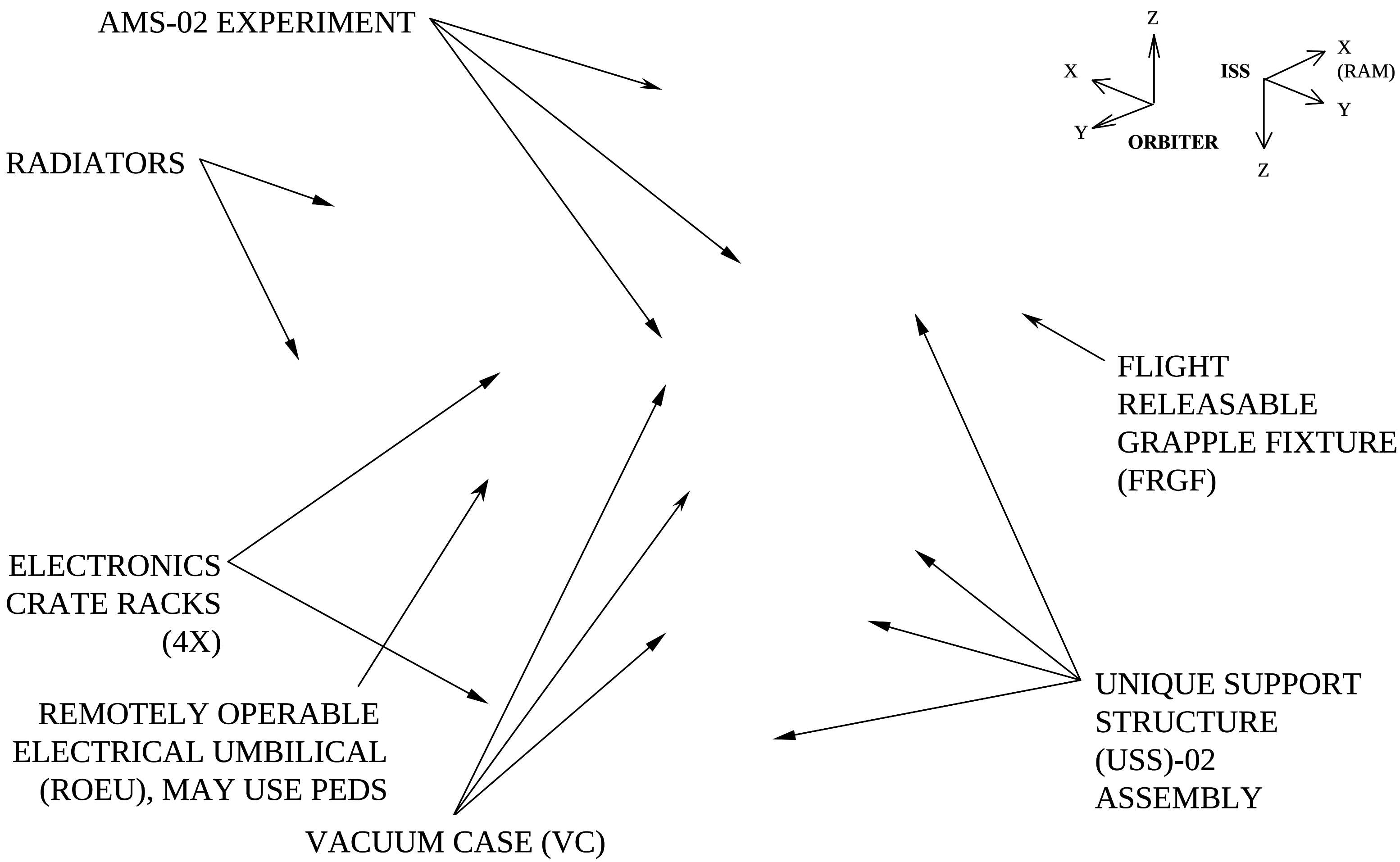


Figure 5.1.1.1 – Vacuum Case as a part of AMS-02 Payload (Sheet 1 of 2)

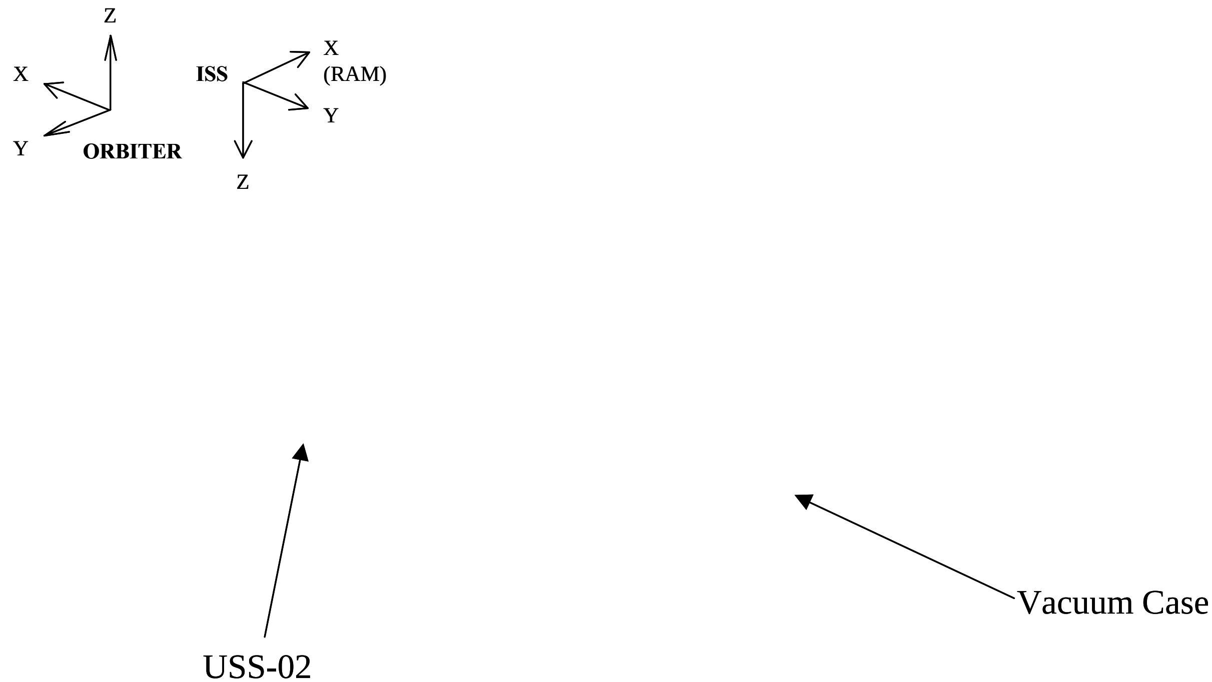


Figure 5.1.1.1 – Vacuum Case as part of AMS-02 Payload (Sheet 2 of 2)

Figure 5.1.1.2 – Vacuum Case Assembly

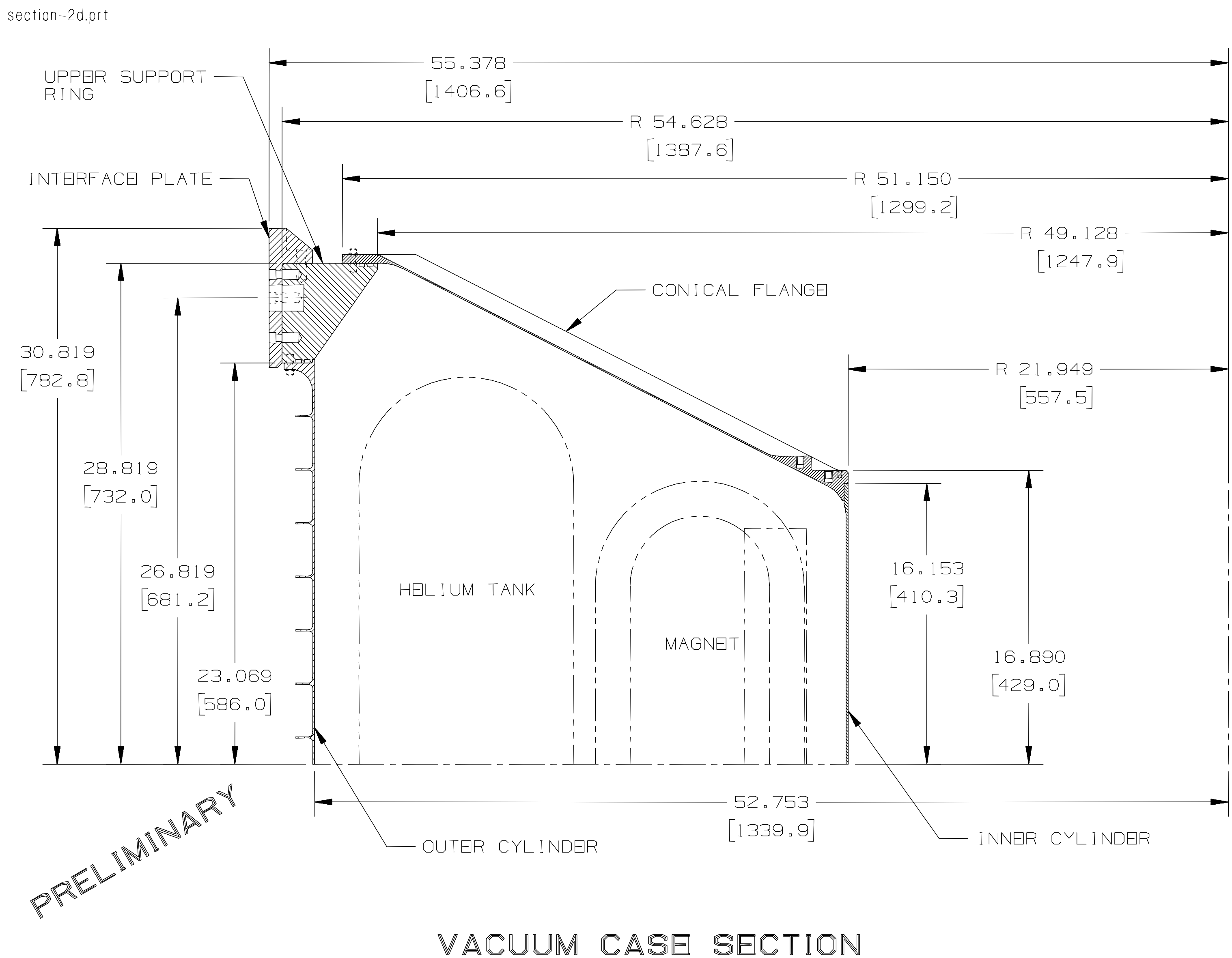


Figure 5.1.1.3 – Vacuum Case Section

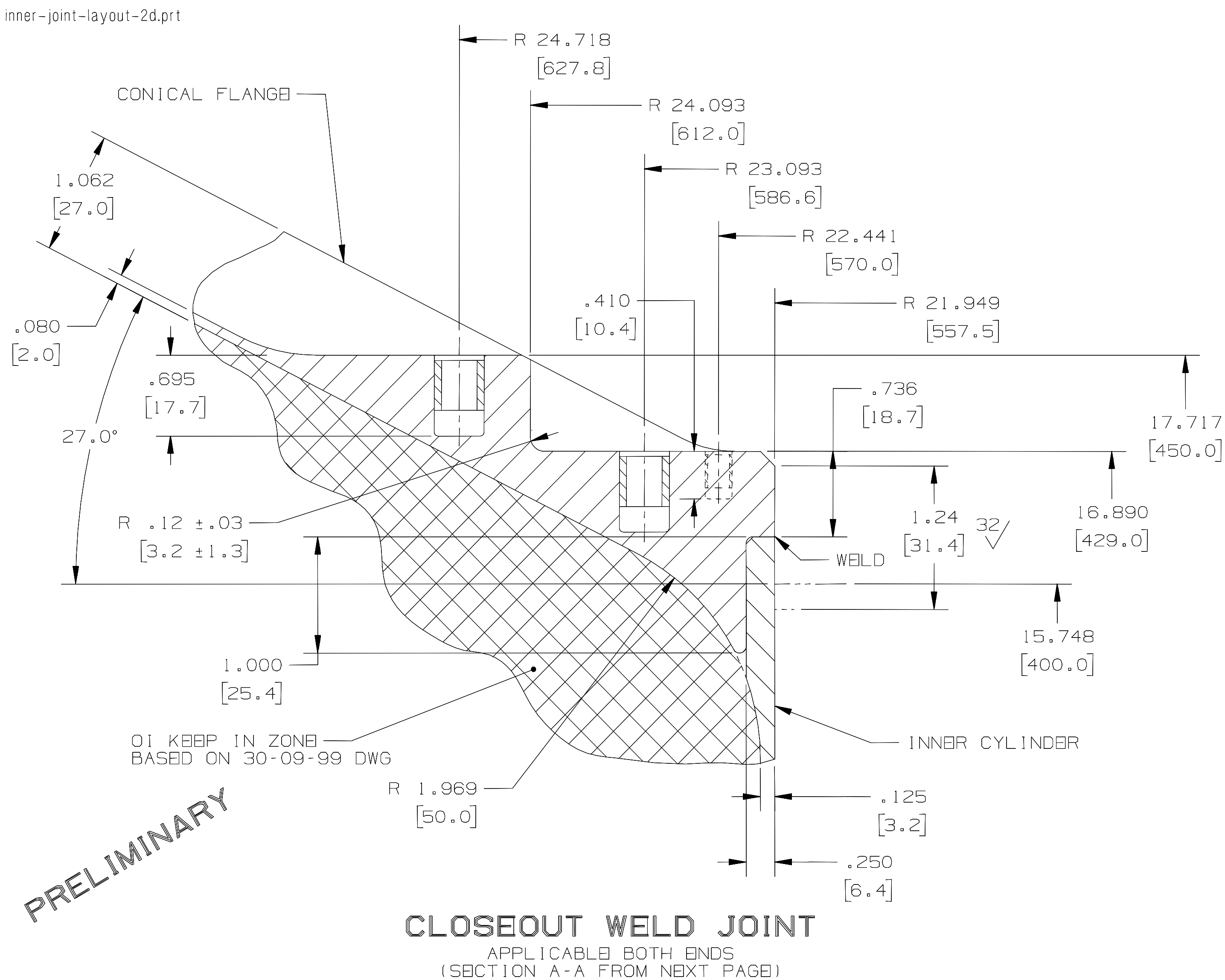


Figure 5.1.1.4 – Close-out Weld Joint

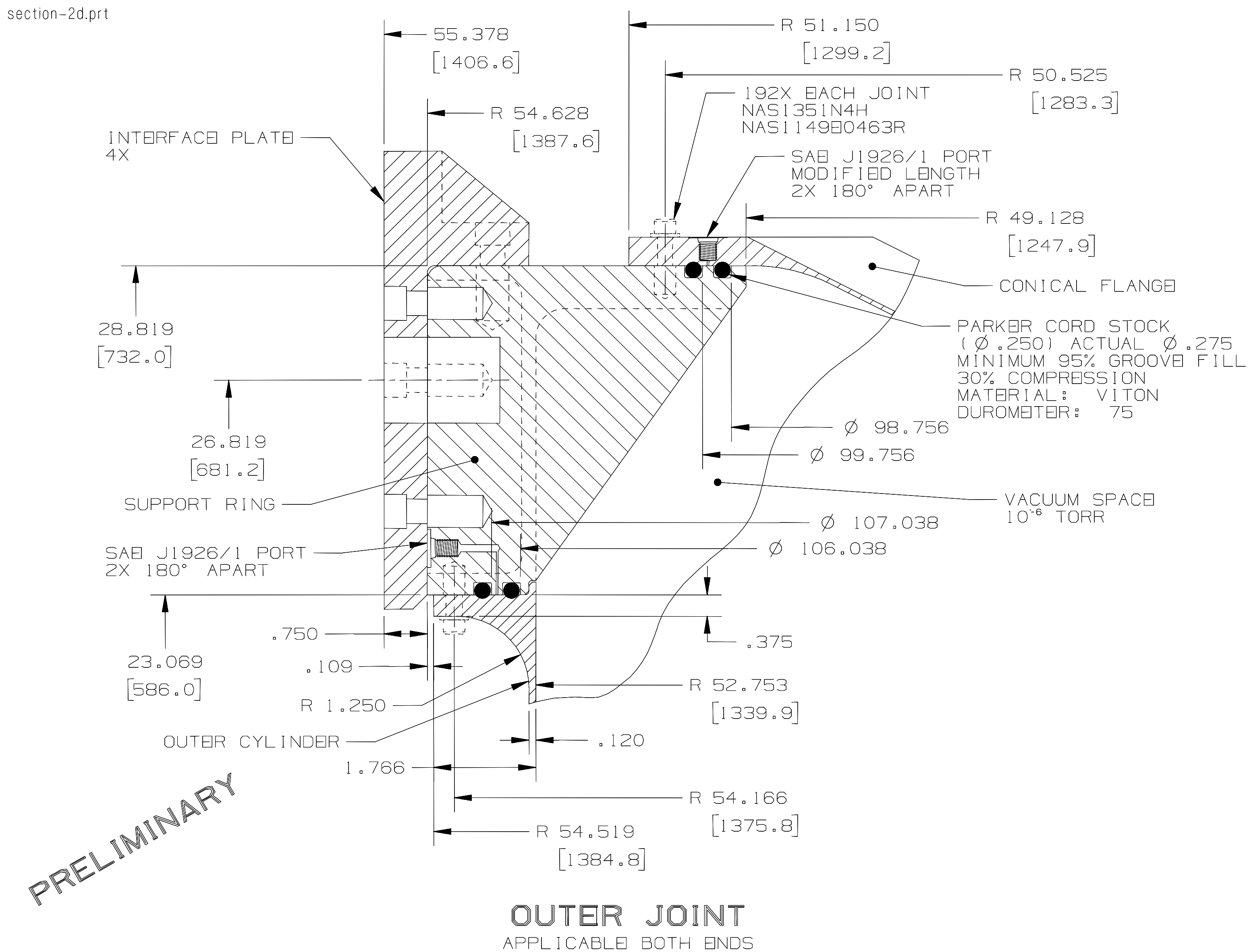


Figure 5.1.1.5 – Outer Joint

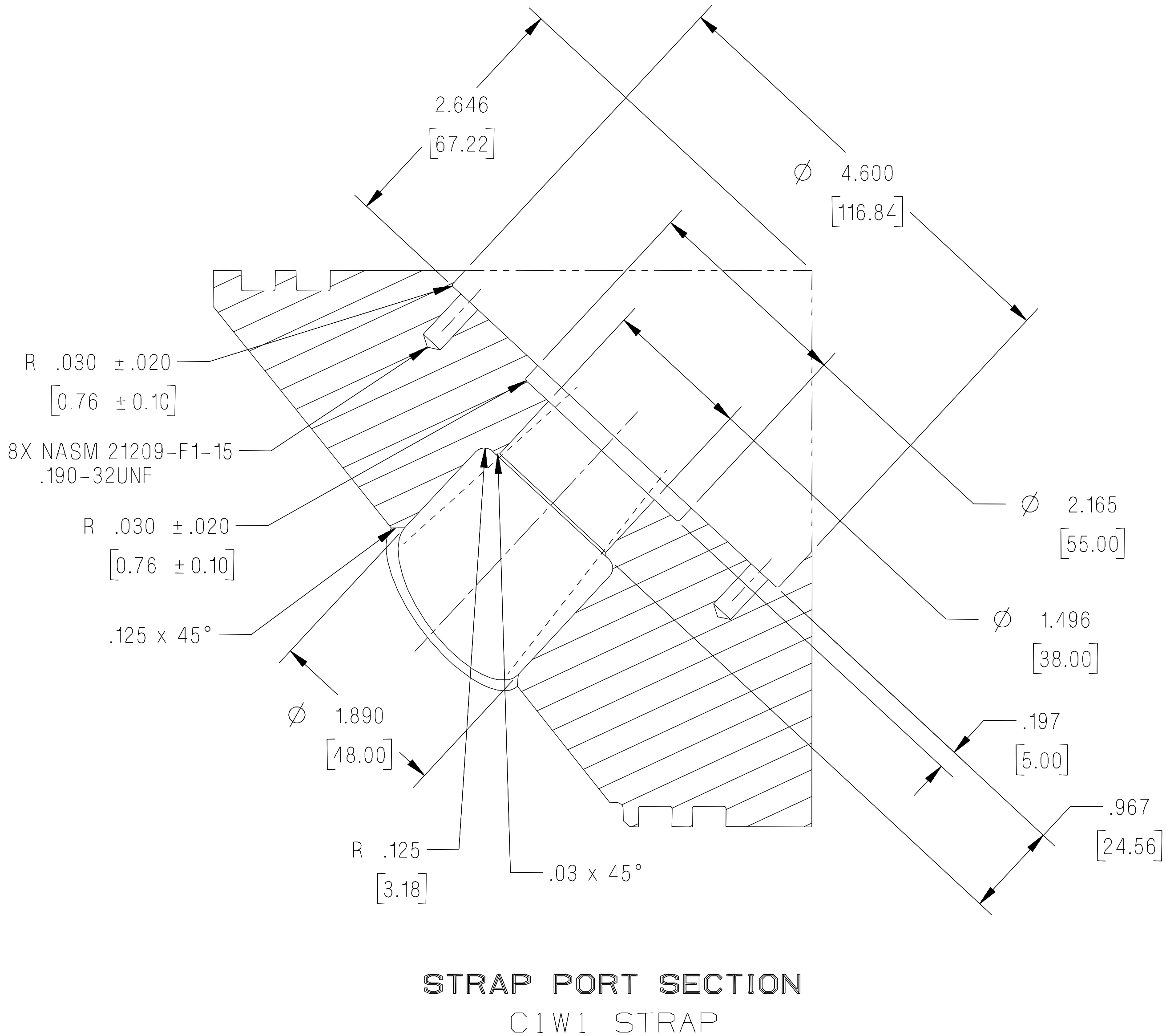


Figure 5.1.1.6 – Strap Port Section

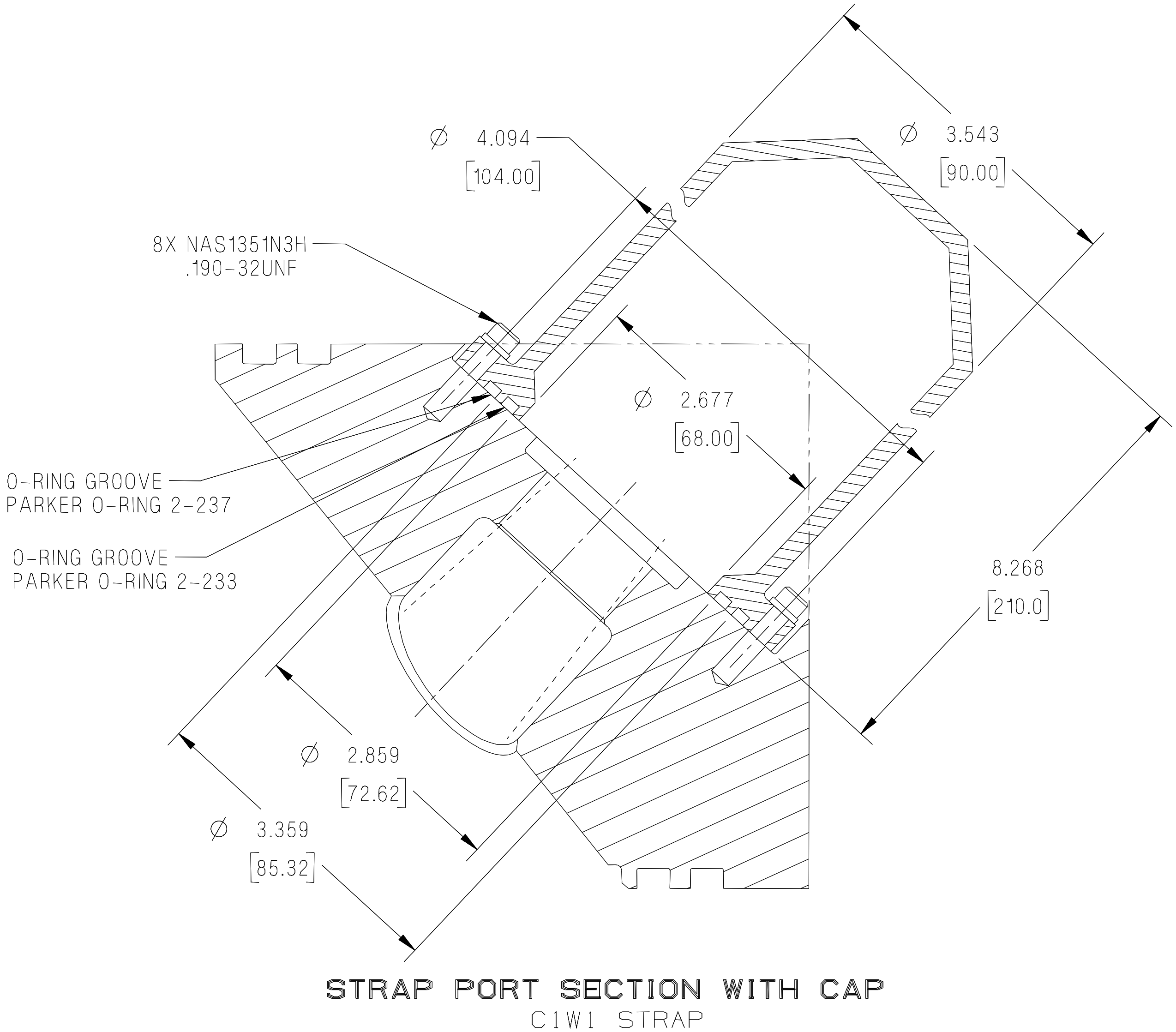


Figure 5.1.1.7 – Strap Port Section With Cap

Removed from Consideration

Figure 5.1.1.8 – 2 inch Port Design

Figure 5.1.1.9 – 4 inch Port Design

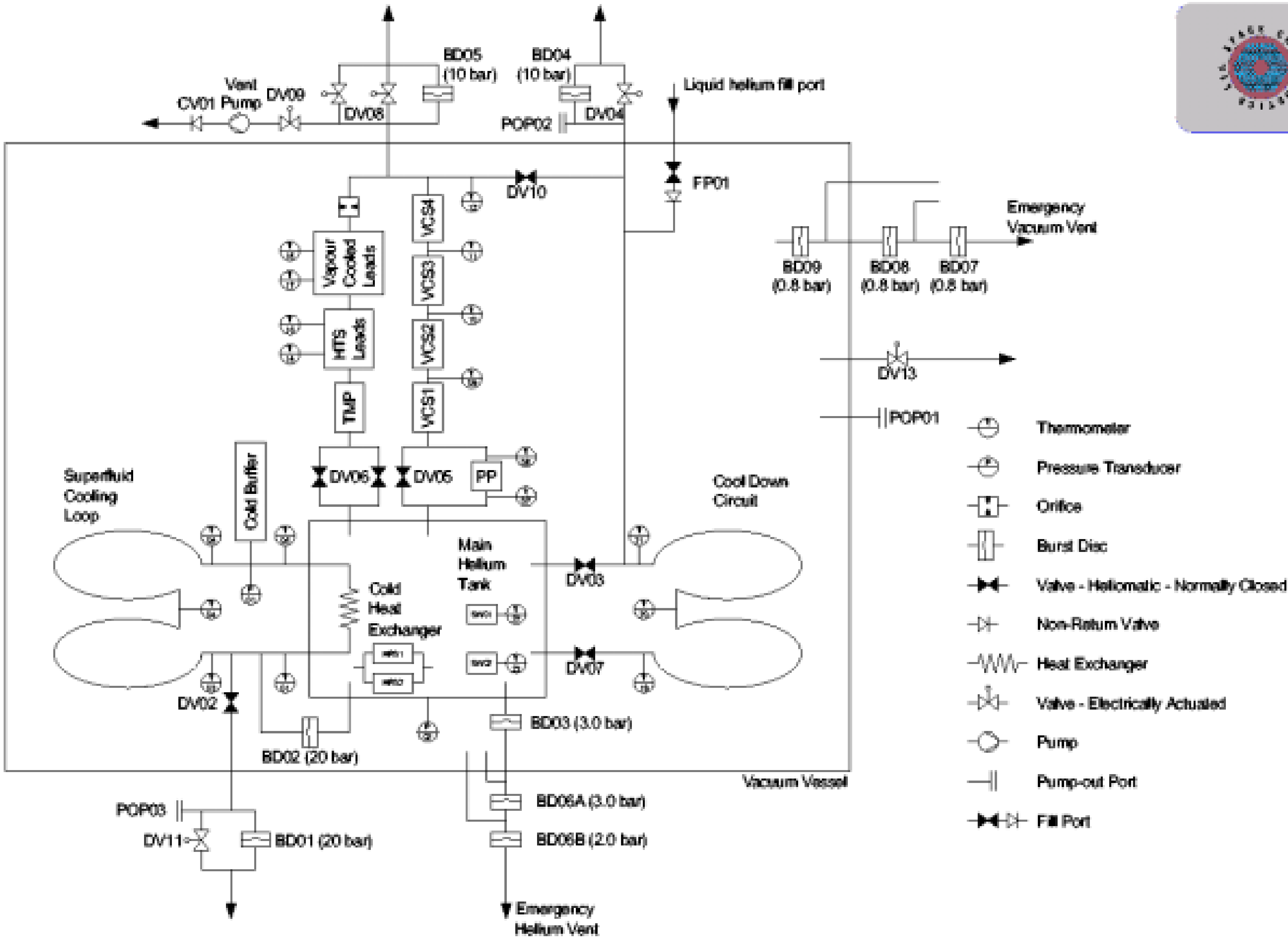
The cryogenic insulation system is composed of 160 layers of superinsulation (MLI) and 4 Vapor Cooled Shields (VCSs). The VCSs will either be a thin aluminum layer or a thin aluminum honeycomb structure.

The cryogenic system also includes valves both internal and external to the VC. All of the valves that are inside the VC are pneumatically operated and are of a design that is similar to one that was flown on the CRISTA SPAS Shuttle mission. All of the valves that are outside of the VC will be electrically activated, but the details for both of these valve systems are TBD. The overall cryogenic schematic is show in Figure 5.1.1.10.

The cryogenic and magnet operations are described in Section 5.1.2. In order to operate the pneumatic valves, a tank of warm helium must be mounted outside of the VC to the USS-02. It will most likely be mounted next to the TRD Gas Supply box. The current proposal is to use a tank that is developed by Arde, Inc. This is the same company that is delivering tanks for the TRD Gas Supply system and has made numerous tanks for other payloads and components of the ISS, STS, and other space vehicles including the X-33. Stainless steel plumbing lines will come out of the warm helium tank and feed two different solenoid control boxes which will be mounted on opposite sides of the VC rings. The solenoid control boxes will be small aluminum boxes that are used to house the various solenoids that will be used to control the gas flow to the pneumatic valves. There will be a small control box for the electrical valves. This box will most likely be located on or inside the Cryomagnet Avionics Box (CAB), which is mounted to the USS-02. The CAB is an aluminum box that is used to house all of the Cryomagnet avionics.

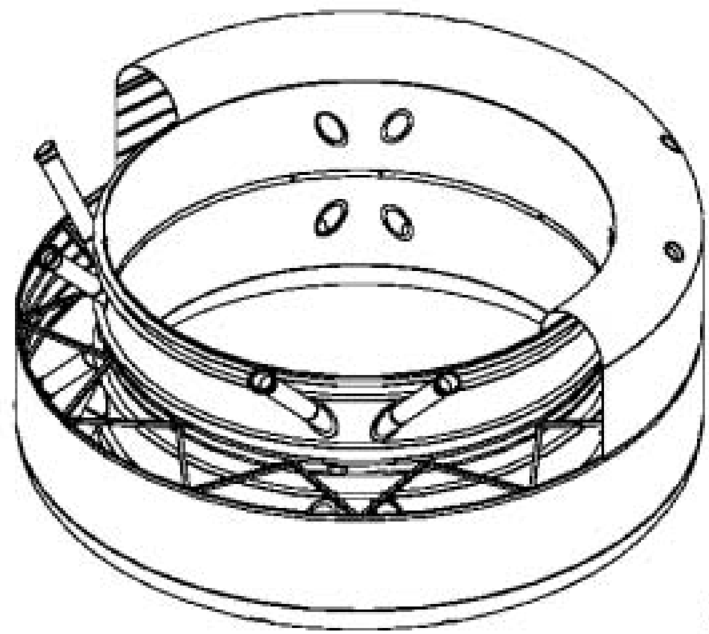
The superfluid helium tank (~2500 liters) will be designed by ETH through a subcontract. It is likely that the tank will be assembled in Switzerland, Taiwan, or England. The toroidal tank design is composed of a central support ring, a rib stiffened inner and outer cylinder, two dome covers, and 16 strap feed-thru tubes (See Figures 5.1.1.11 & 5.1.1.12). The tank will also have 3 burst disks in series, which will vent outside the VC, for emergency helium venting. The tank is made of either aluminum 5083-T0, 6061-T6, or 2219-T87 and has all welded interfaces. The system is built up in two complete halves (top and bottom) that are welded to the center support ring (washer shaped). This allows for complete radiographic inspection prior to the last closeout welds. The inner cylinder has a radius of ~37.8 inches (953 mm), and the outer cylinder has a radius of ~50.8 inches (1283 mm). The height of the helium tank is ~47 inches (1194 mm).

The superfluid helium tank attaches directly to the magnet support structure at various locations around the center support ring through a system of shear pins. The arrangement allows for differential thermal expansion of the tank relative to the magnet. The tank operating temperature is 1.8 degrees Kelvin. The magnet will be at a very similar temperature, but could be slightly higher.



Note: Redundant Pressure and Temperature Gauges not shown.

Figure 5.1.1.10 – Cryogenic Schematic (Updated)



ISOMETRIC VIEW SHOWING PART
SECTION OF HELIUM VESSEL

HELIUM VESSEL FOR AMS - 02

SHEET 1

Figure 5.1.1.1.11 – Superfluid Helium Vessel for AMS-02 (Sheet 1 of 2)

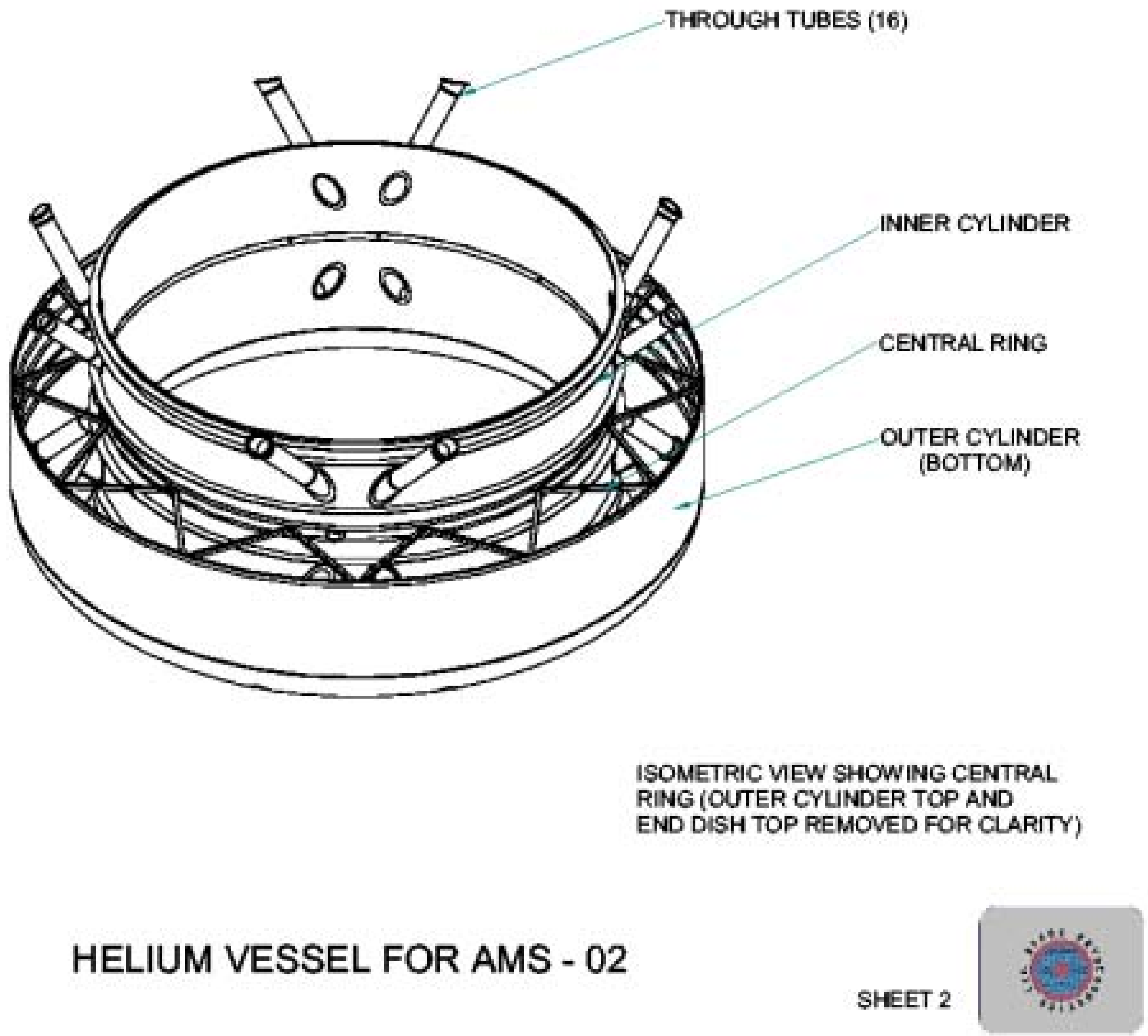


Figure 5.1.1.12 – Superfluid Helium Vessel for AMS-02 (Sheet 2 of 2)

The magnet itself is composed of 12 racetrack coils and two dipole coils (See Figure 5.1.1.13). The coils are supported by an aluminum 6061-T6 island. The niobium/titanium, copper stabilized conductor is co-extruded with an aluminum stabilizer and the resulting conductor is wrapped around the aluminum island. The islands of the 12 racetrack coils bolt and pin together to make 2 quarters of the magnet. The remaining 2 quarters are the dipole coils, which have a similar design, but are much larger. The support structure of these sections is called the racetrack endframe tie plates. At the 4 corners of the magnet, and at the ends of the racetrack endframe tie plates is where the magnet support straps attach. This is also the location where the superfluid helium tank attaches to the magnet, in the z-axis.

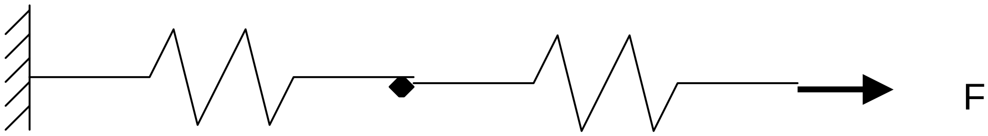
The magnet support system key design driver is to allow a minimal heat leak into the magnet. The strap system is designed to minimize the heat load, absorb the thermal contraction, fit within the very tight geometric constraints, require very little pre-load so as to avoid long duration loading concerns, and maintain tension during all loading phases. The non-linear straps and their operation are shown in Figures 5.1.1.14 thru 5.1.1.17. The system is called the Tension Only Passive Orbital Disconnect System (TOPODS). The system is composed of three different composite strap systems that create a non-linear strap system. The system is defined by three distinct stiffness regions. The stiffness characteristics allow the strap system to carry more load during launch and landing then they will under unloaded cases. In the warm condition, both ends of the strap system are at room temperature. Under this condition, the launch/landing portion of the strap is engaged and the strap will carry more load. In the cold condition with the system in the unloaded configuration (i.e., No launch/landing loads, but does have nominal preload), one end at room temperature (~300 degrees Kelvin) on the vacuum case, the other end at 1.8 degrees Kelvin, the launch/landing strap is disengaged. As the launch/landing loads are applied, the launch/landing strap becomes engaged and carries more load. The straps connect the vacuum case at the upper and lower ring. They go through the strap feed-thru tubes in the superfluid helium tank and then attach to the magnet. The pin ended straps never touch the strap feed-thru tubes on the superfluid helium tank.

The strap design is based on the following:

Assumptions:

- The composite strap (either the on-orbit tie or a combination of the on-orbit tie and the launch/landing tie) is attached to the Composite Belleville washer forming the equivalent of two linear springs in series.
- The stiffness of the composite strap is significantly larger than the stiffness of the Belleville washer
- Belleville washer stiffness = k_1
- On-Orbit strap stiffness = k_2
- Launch/Landing strap stiffness = k_3

(1) When load is initially applied to the combined system, both the strap and the Belleville will have a deflection due to the applied load.



$$\delta_s = F/k_s \qquad \delta_b = F/k_b$$

The combined stiffness for two springs in series is equal to the applied load divided by the sum of the total deflection:

$$k_c = \frac{F}{\frac{F}{k_b} + \frac{F}{k_s}} = \frac{F}{\frac{Fk_b + Fk_s}{k_b k_s}} = \frac{k_b k_s}{k_b + k_s}$$

Since k_s >> k_b, $k_c \approx k_b$ i.e. the equivalent stiffness of the combined system is approximately the stiffness of the Belleville.

(2) When sufficient load is applied to cause the Belleville to bottom out, the stiffness of the Belleville effectively becomes much greater than the stiffness of the strap. (The Belleville is now a flat disk in compression.)

The combined stiffness for the system with **k_b >> k_s** is given by the same equation as previously shown above.

$$k_c = \frac{k_b k_s}{k_b + k_s}$$

However, now the resulting stiffness from this equation is $k_c \approx k_s$ i.e. the combined system stiffness is approximately the stiffness of the strap alone.

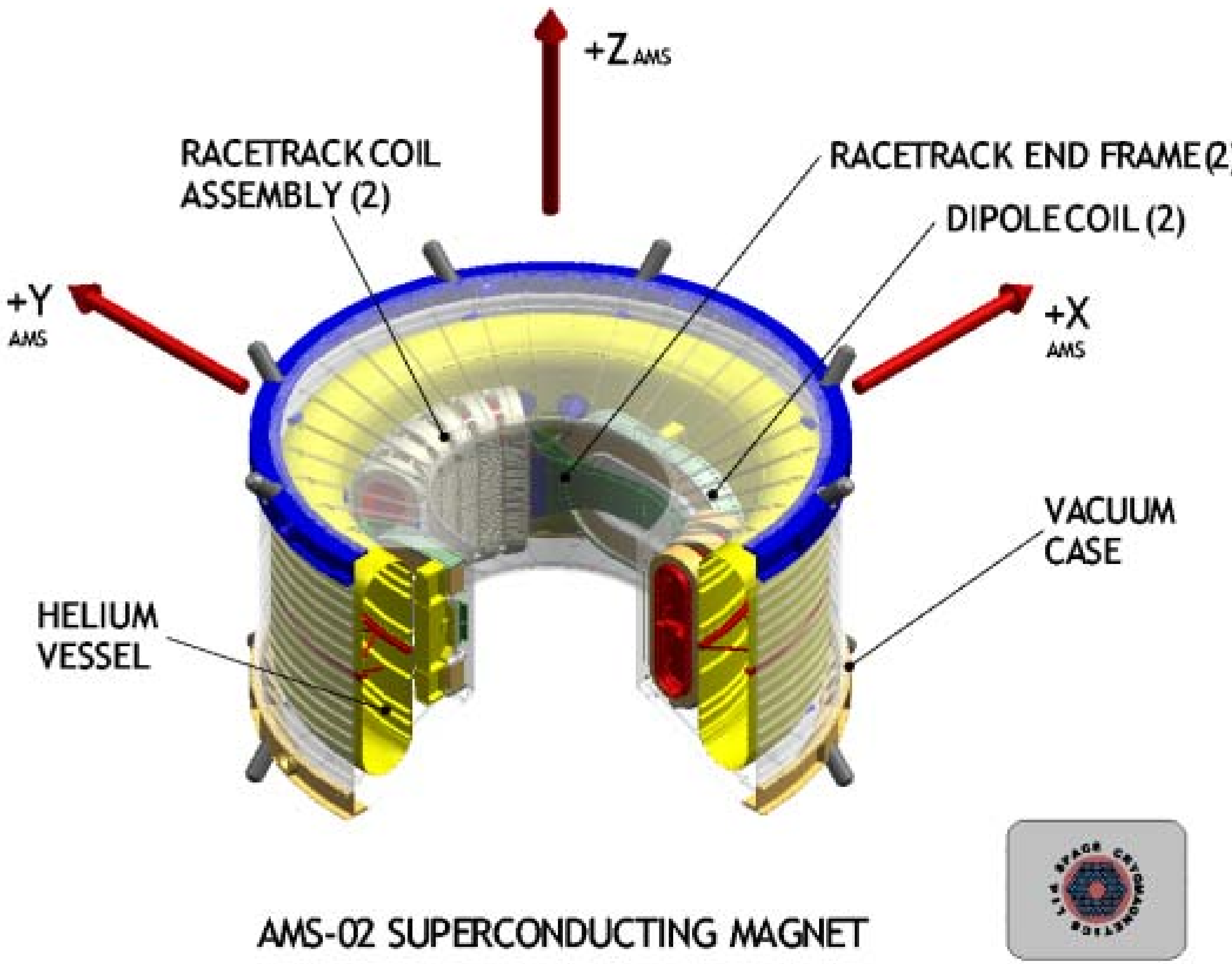


Figure 5.1.1.13 – AMS-02 Superconducting Magnet Layout (Updated)

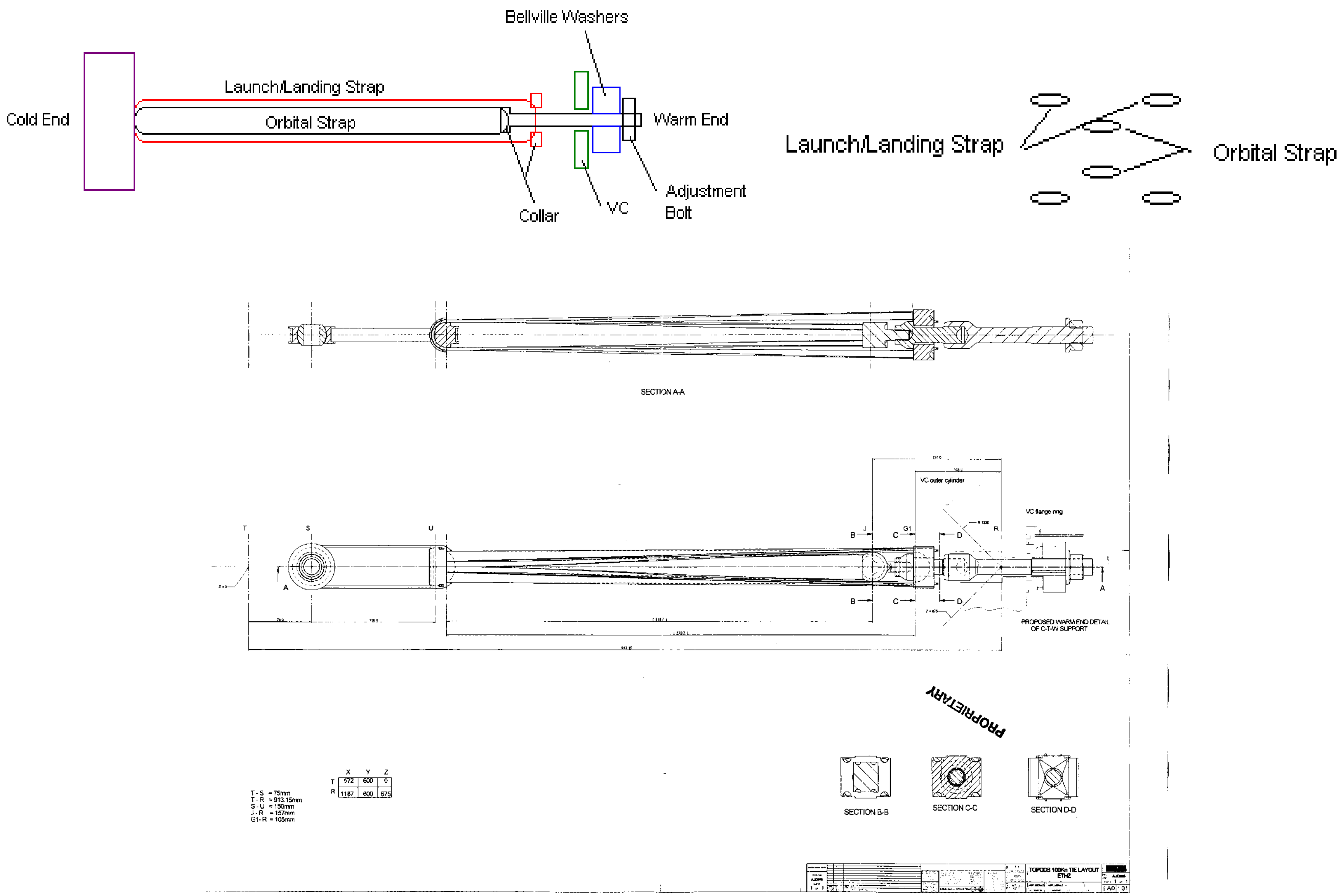


Figure 5.1.1.14 – Cryosystem Support Strap

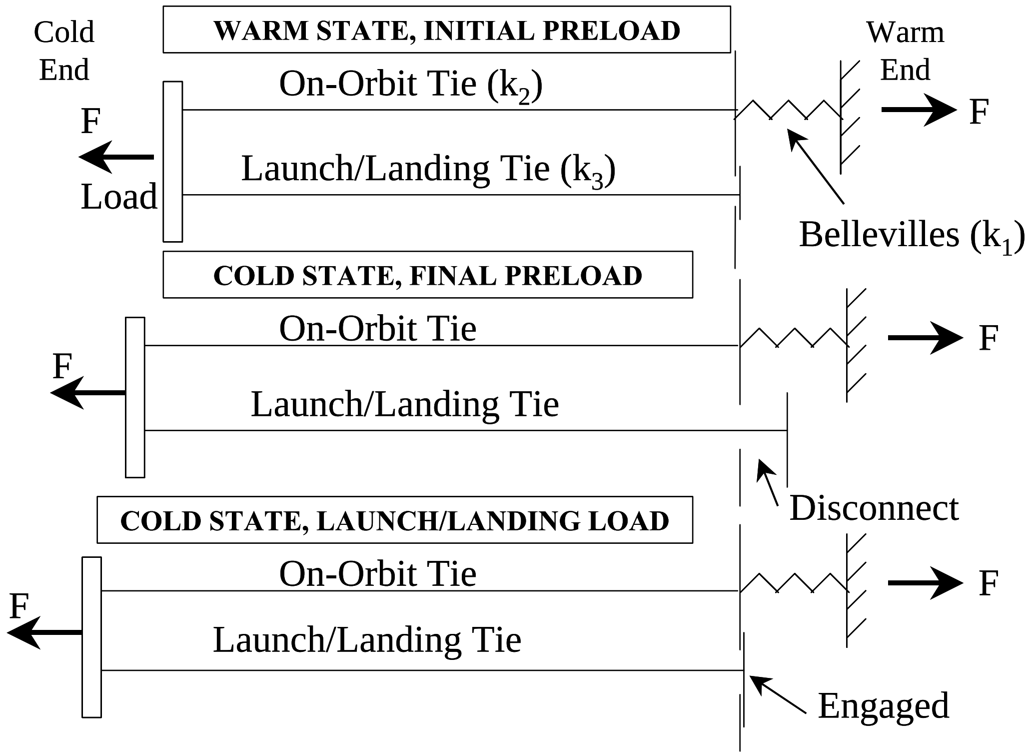


Figure 5.1.1.15 – Cryosystem Support Strap Configuration and Functional Diagram

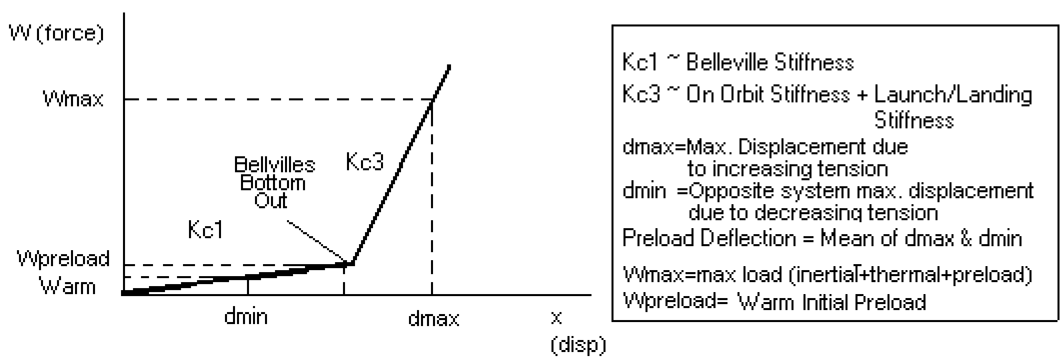


Figure 5.1.1.16 – Cryosystem Support Strap Warm State

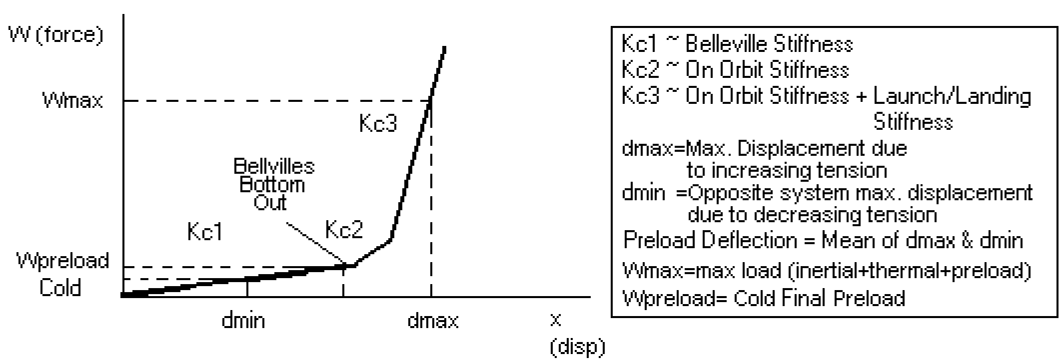


Figure 5.1.1.17 – Cryosystem Support Strap Cold State

5.1.2 Cryomag Operation

This section describes the operating modes and control for the cryogenic system concept for the AMS-02 superconducting magnet system. Sections 5.1.2.2 through 5.1.2.5 are prelaunch ground operations.

Due to some recent venting analyses performed by AMS-02 and STS Integration, the emergency venting of the helium from the SFHe tank as a direct impact on the prelaunch operations.

Although no credible scenarios could be found that would cause an emergency vent in the Orbiter Payload Bay during launch/landing, AMS-02 and Shuttle Integration have assessed the AMS-02 emergency vent rate assuming a complete loss of vacuum on the ground. Based on the conservative AMS-02 and STS Integration Helium Venting analyses that have been performed to date:

- The only time that there is an overpressure issue with the Orbiter Payload Bay is if the SFHe tank burst disk ruptures between ~T+30 & ~T+60 seconds.
- There is no landing or other scenarios that can cause an overpressure of the Payload Bay.
- Analysis predicts that it will take ~180-220 seconds after a total loss of vacuum before the SFHe tank burst disk ruptures.
- Launch Commit Criteria (up to T-31 seconds) will be in place to show that there is no loss of vacuum during this time frame (~T-190 to ~T-120 seconds). Loss of vacuum after T-31 would not provide the time required to cause the helium vent during the critical ascent region of T+30 to T+60 seconds (would be T+150 seconds)
- Venting tests will confirm both the time to SFHe tank burst disk rupture and the Helium flow rate.
- Test data will be compared to analysis data to confirm these predicted results.
- The thermal issue due to helium venting is still open and in work.
- STS Integration agrees with this assessment.

NOTE: These operations are for mission success only and are NOT safety critical. All pressure and temperature sensors will be redundant to ensure accurate measurements. Both the VC and the SFHe tank pressure and temperature will be monitored.

5.1.2.1 Abbreviations and Some Limited Definitions

BD Bursting Disc

CBV Cold Buffer Volume

CDC Cool Down Circuit

CHX Cold Heat Exchanger

DV Digital (on/off) valve

HTS High Temperature Superconductor – The HTS leads consists of silver-stabilized high temperature superconductor with stainless steel shunts. The superconductor itself is a compound of bismuth, strontium, calcium, copper and oxygen generally referred to as BSCCO-2223. During magnet charging, the leads are cooled by helium supplied by the TMP, and no heat is generated because of the superconductor. If the coolant flow fails at a critical moment, the leads and shunts together have sufficient thermal inertia to allow the magnet to be discharged safely.

MHT Main Helium Tank – Also designated Superfluid Helium (SFHe) Tank

PP Porous Plug – The unique properties of SFHe are utilized on-orbit to allow the separation of the liquid and vapor of SFHe. A porous plug is typically a small cylinder shape piece of porous metal (often made of stainless steel).

SCL Superfluid Cooling Loop

TMP Thermo-Mechanical Pump – This pump uses the unique nature of superfluid helium to generate hydraulic head simply by the application of heat. The pump consists of a very fine filter and a heater element. The heater is positioned on the downstream side of the filter. When heated, it causes helium to flow through the filter. This will only operate if the helium is below the lambda transition temperature (2.17 K). The maximum pressure which can be generated is around 10.2 psi (0.7 bar). Pumps of this type (but larger than the AMS version) were flown on SHOOT. The AMS pump is currently under development.

VCL Vapor Cooled Leads – Simple brass or copper conductors which carry current between the warm bus bars (outside the vacuum case) and the HTS leads. These are also cooled by the helium supplied by the TMP.

5.1.2.2 Cool Down 300 K to 80 K

(i) Initial cool down will use helium gas which is pre-cooled in a liquid nitrogen heat exchanger. This is to reduce the possibility of contaminants such as air or water entering the helium system.

(ii) Connect a supply of warm gas to the HP FILL interface. Open DV02 and DV11 and maintain the pressure in the SCL at around 58 psia (4 bara) throughout this stage of cool down.

(iii) Connect the pre-cooled helium supply to the LP FILL/VENT interface. Open DV04, DV07, DV05 and DV08. Close DV03, DV06, DV9, DV10 and DV12. The cooled helium will now flow through the CDC, into the MHT and out through the shields.

(iv) Steadily reduce the temperature of the helium supply until the system reaches 80 K.

5.1.2.3 Cool Down 80 K to 4.2 K

(i) Maintain the pressure in the SCL using the warm gas supply.

(ii) Position the system so that the outlet from the MHT into the shield circuit via DV05 is at the highest possible position.

(iii) Close DV03, DV06, DV07, DV09 and DV12. Open DV04, DV05, DV08 and DV10. Connect a liquid helium supply to the LP FILL/VENT interface. Siphon liquid helium into the system: the helium will bypass the coils and the MHT and flow through DV10. When TT12 indicates that the warm helium from the transfer line has all been vented, open DV03 and DV07, and close DV10. The MHT and the CDC will fill with liquid helium: vapor will be vented through the radiation shields.

5.1.2.4 Cool Down 4.2 K to 1.85 K

(i) Allow the system to fill with liquid helium.

(ii) Increase the pressure in the SCL to 142 psia (9.8 bara).

(iii) Disconnect the warm helium supply from the HP FILL interface.

(iv) Close DV02 and DV11. Evacuate the HP FILL line. The high-pressure circuit is now isolated.

(v) Connect the cold helium supply to the LP FILL/VENT interface: this may be a supply of normal liquid helium with a throttling valve, or possibly a supply of refrigerated superfluid at a pressure of 14.5 psia (1 bara) or more.

(vi) Open DV09 and connect the vent pump to the outlet.

(vii) Ensure that DV03, DV04, DV05, DV07 and DV09 are all open. Ensure that DV06, DV10 and DV12 are all closed.

(viii) Reduce the pressure in the MHT by pumping on the shield interface, simultaneously filling through DV04 and DV03.

(ix) Wait for the system to fill with superfluid and become stable at a pressure below 0.29 psia (20 mbara). Continue pumping the shield interface.

(x) Close DV05. Check the operation of the porous plug phase separator.

(xi) Close DV07.

(xii) Close DV03.

(xiii) Disconnect the cold helium supply from the LP FILL/VENT interface.

(xiv) Ensure that DV03, DV07, DV10 and DV12 are all closed. With DV04 open, connect a vacuum pump to the LP FILL/VENT interface and evacuate the LP FILL line. This will also evacuate the coils cool down circuit. Close DV04 and remove the pump.

(xv) The system should now be operating in normal steady state conditions.

5.1.2.5 Steady State Operation

(i) Apart from the lack of gravity, steady state operation in space should be the same as on the ground.

(ii) DV02 and DV11 are closed, with the HP FILL line evacuated. This isolates the high pressure cooling circuit.

(iii) The SCL is full of supercritical superfluid helium at about 1.85 K and 43.5 psia (3 bara). The magnet heat load is intercepted by the helium in the SCL and transferred to the MHT by Gorter-Mellink conduction.

(iv) DV03, DV04, DV07 and DV12 are closed, with the space enclosed between them evacuated. This space includes the CDC.

(v) DV05 and DV06 are both closed. The only outlet from the MHT is therefore through the porous plug phase separator and the shields.

(vi) There is no flow through the current leads.

(vii) The shield flow is not controlled, but depends only on the boil-off from the MHT. The equilibrium pressure in the MHT depends on the heat load and the size and geometry of the phase separator and vent line.

5.1.2.6 Launch

(i) Before launch the system should be operating stably, with a vacuum pump on the current lead and shield vent lines outlet maintaining the low pressure.

(ii) At the last possible minute before launch, the pump and cryocoolers are switched off. All valves on the system are closed.

(iii) During launch, when the exterior pressure is lower than the MHT pressure, but while there is still acceleration keeping the helium at the bottom of the tank, DV08 is opened to vent the MHT to space through the shields. This operation will require momentary power during ascent (see Section 6.0). (This is to avoid "Castles Catastrophe", a thermal effect which pumps all the superfluid helium out of the MHT.)

(iv) As soon after launch as possible, the cryocoolers are turned on. DV04 and DV11 are opened to vent the interface lines to space. DV13 is opened to vent the vacuum to space. These valves can be opened at any time up to several days into the mission.

(v) After a transient (which may last a few days), the cryogenic system should operate stably.

5.1.2.7 Charge/Discharge

(i) To change the current in the magnet (either up or down) it is necessary first to cool the current leads.

(ii) Open DV06.

(iii) Apply heat to the TMP in the current lead circuit. This will pump helium from the MHT to cool the HTS bus bars and the resistive current leads.

(iv) Monitor the temperature TT17 of the current leads. When it has fallen to 75 K the magnet can be charged or discharged.

(v) When the steady state current has been reached, the superconducting switch closed and the power supply switched off, DV06 is closed and the TMP heater switched off.

5.1.2.8 Quench in Space

(i) If the magnet quenches from full field, the average temperature rises to about 65 K.

(ii) The pressure in the SCL rises, but the helium expands into the CBV and is not discharged into the MHT.

(iii) DV07 and DV04 are opened to allow helium from the MHT to flow through the CDC and re-cool the coils. Additional heat will be transferred from the CBV and SCL into the MHT during this time. The time allowed to take this action depends on the amount of helium left in the MHT, but a few 10s of minutes should be sufficient.

(iv) When the coil temperature is sufficiently low, the SCL will begin to operate again. DV07 is closed; the helium remaining in the CDC is vented to space (there is no need to close DV04 again). The system now returns to steady state operation.

5.1.2.9 Launch Abort

(i) If the launch is aborted and the system remains on the pad, it may be necessary to take some actions to prevent eventual venting of the MHT through BD03, BD06A and BD06B. However, there will be several hours available before any intervention is required.

(ii) The vent pump will have been switched off for launch. If this can be restarted within a few hours, no further actions will be necessary.

(iii) The pressure in the MHT can be monitored using pressure transducer PT02. If the pressure in the MHT rises above atmospheric pressure, open valves DV05, DV10 and DV12. This will allow the MHT to vent, if required, throughCV02 to the atmosphere. This action is also appropriate if it is not possible to re-start the vent pump.

5.1.2.10 Emergency Return from Space

(i) Close all valves. This will bottle up the helium inside the MHT and prevent air or ice being sucked back into the system.

(ii) When the pressure inside the MHT exceeds 21.8 psi (1.5 bar), open DV12 to allow the (fully normal) helium to vent through the cool down circuit.

5.1.2.11 Quench on the Ground

(i) A quench on the ground will be similar to a quench in space. The only difference is that it will be necessary to attach a vacuum pump to the LP FILL/VENT interface to allow the helium through the CDC to re-cool the coils. DV04 has to be closed after DV07 once the magnet is cold.

5.1.2.12 Loss of Vacuum

(i) Loss of vacuum can only occur on the ground, at the beginning of launch, or at the end of landing.

(ii) The SCL vents into the MHT through BD02.

(iii) The MHT vents outside the vacuum vessel through BD03, BD06A and BD06B.

(iv) The vacuum space vents to atmosphere if necessary through BD09, BD08 and BD07.

(v) Loss of vacuum cannot be recovered without intervention on the ground.

5.1.2.13 Warm up in Space – Natural

(i) When the system runs out of superfluid helium, it will begin to warm up, eventually reaching ambient temperature.

(ii) The pressure in the SCL will gradually increase. DV02 and DV11 may be opened, in which case the SCL will vent to space. If these valves are not opened, the SCL will vent into the MHT through BD02.

(iii) The pressure in the MHT will gradually increase. DV07 may be opened to vent the MHT to space. If DV07 is not opened the MHT will eventually vent to space through the burst discs BD03, BD06A and BD06B.

(iv) Internal parts of the system will probably remain fairly cold (below 200 K) for several months. If the magnet is returned to Earth without closing the vent valves during this time condensation will form inside the helium vessels. This will not be a hazard.

5.1.2.14 Warm Up on the Ground

(i) Fit a non-return valve to the HP FILL interface. This is to allow helium to vent from the SCL without sucking air back into the helium system.

(ii) Open DV02 and DV11 to allow helium out of the SCL.

(iii) Open DV03 and DV12 to allow helium out of the MHT through the non-return valve CV02.

(iv) Switch off any mechanical coolers.

(v) Close DV08 and DV09. Open DV05, DV06 and DV10. The increased heat load from the radiation shields will increase the warm up rate.

(vi) Power any available heaters.

(vii) Once all parts of the system are above 80 K, degrade the system vacuum using dry nitrogen gas.

5.2 UNIQUE SUPPORT STRUCTURE-02 (USS-02)

The USS-02 is used to support the AMS-02 cryomagnet and detectors and to interface the entire AMS-02 Experiment with the Space Shuttle Orbiter and ISS. The vacuum case is also an integral part of the USS-02. The USS-02 is comprised of the following five subassemblies: (1) Upper USS-02 Assembly, (2) Vacuum Case Assembly, (3) Lower USS-02 Assembly, (4) Keel Assembly, and (5) Payload Attach System (PAS)/Umbilical Mechanism Assembly (UMA) Assemblies (See Figures 5.2.1 - 5.2.2).

The USS-02 primary members consist of 4 inch (101.6 mm) square tubing with 0.25 inch (6.4 mm) walls made from 7075-T73511 extruded aluminum tubing, 6.5 x 6.0 inch (165.1 x 152.4 mm) square tubing with 0.25 inch (6.4 mm) walls made from 7050-T7451 aluminum plate, and 5.0 x 5.5 inch (127 x 139.7 mm) tubing with 0.25 inch (6.4 mm) walls made from 7050-T7451 aluminum plate. The tubes are connected with machined aluminum 7050-T7451 fittings and gussets, and they are fastened with rivets and bolts. The USS-02 attaches to the Space Shuttle Orbiter with four longeron trunnions and one keel trunnion. The AMS-02 payload will be attached to the ISS using the Payload Attach System (PAS). The PAS hardware on the AMS-02 is the passive half and consists of three guide pins and a capture bar.

Several AMS-02 components are mounted to the USS-02. These components include: the SRD, TRD, TRD gas supply system, TOF, RICH, ECAL, electronics crates, RICH electronics, ECAL electronics, Cryo Avionics Box (CAB), cryomag rectifiers, electrical cables and components of the Thermal Control System (TCS). The Space Shuttle Program (SSP) provided hardware that will be attached to the USS-02 include: Flight Releasable Grapple Fixture (FRGF) and Remotely Operated Electrical Umbilical (ROEU) or Passive Electrical Disconnect System (PEDS). The ISS Program provided hardware that will be attached to the USS-02 include: Power Video Grapple Fixture (PVGF), passive Payload Attach System (PAS), passive Umbilical Mechanism Assembly (UMA), Berthing Cues System (BCS), and Meteoroid and Orbital Debris (MOD) shields. Both the passive PAS and the MOD are designed and built by NASA/LM Mission Management Office.

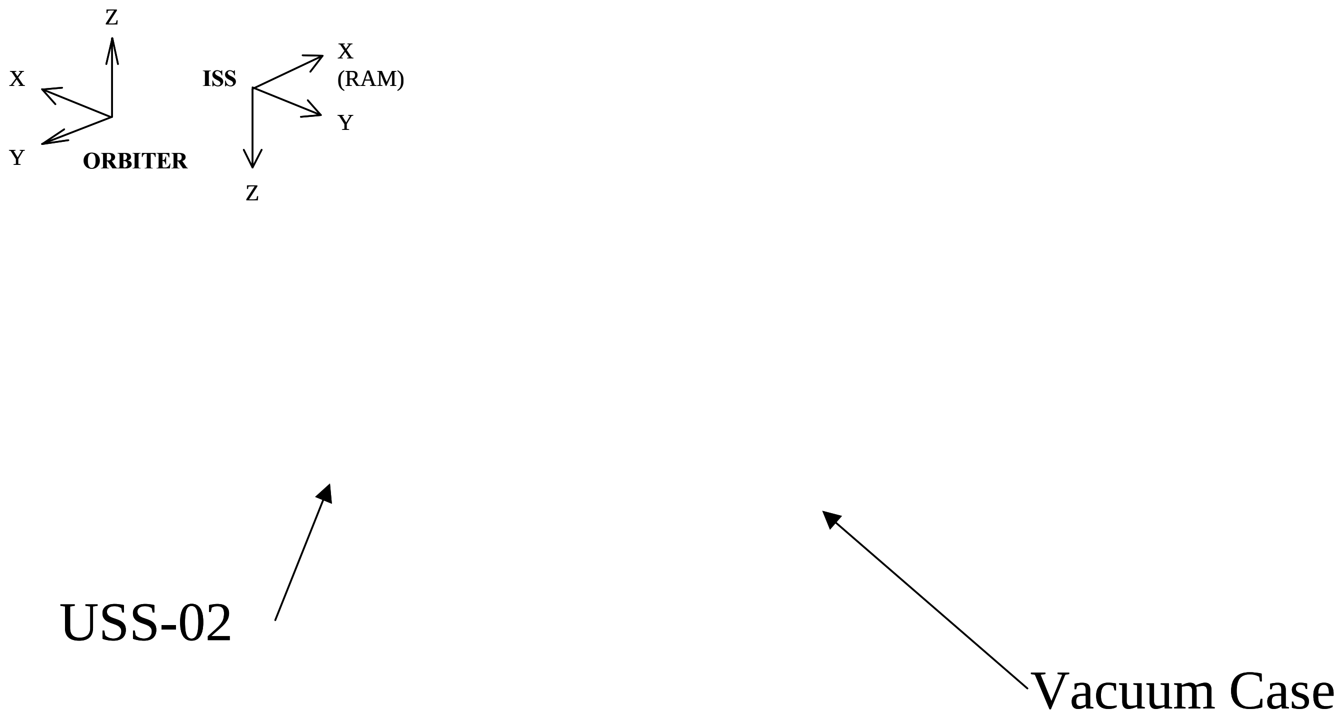


Figure 5.2.1 – Unique Support Structure – 02 (Sheet 1 of 2)

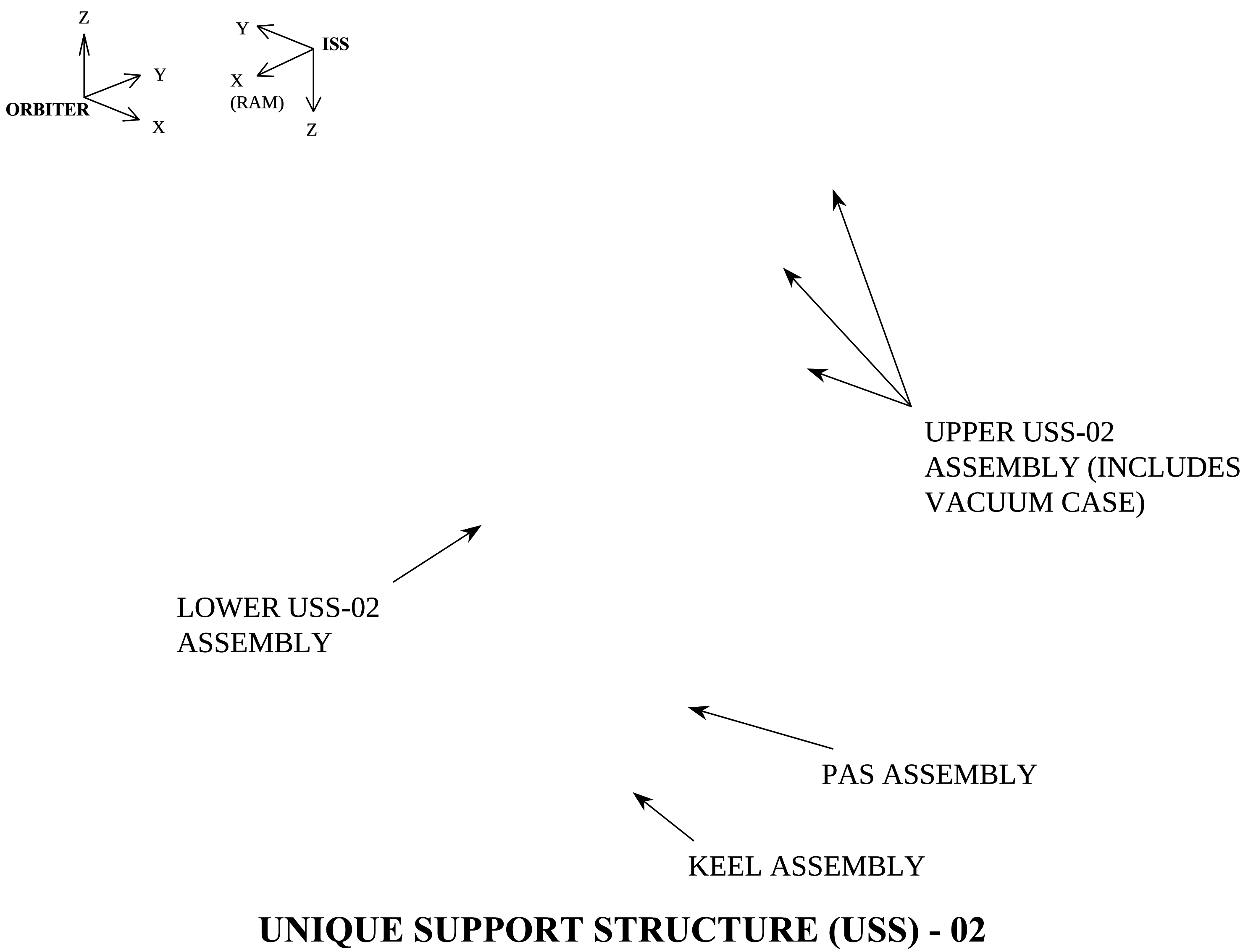


Figure 5.2.2 – Unique Support Structure – 02 (Sheet 2 of 2)

5.3 SYNCHROTRON RADIATION DETECTOR (SRD)

The SRD is located on the top of the experiment stack and attaches to the top of the TRD support at 4 locations. The SRD is approximately 100 x 100 x 4 inches (2540 x 2540 x 101.6 mm) and weighs about 441 lbs (200 Kg). The SRD detector is supported by a large honeycomb panel. The panel will most likely have carbon fiber face sheets and an aluminum core.

The SRD is a particle physics experiment with different detector materials configurations to measure photons, X-rays and low energy charged particles. The SRD is able to extend the accessible energy range beyond 400 GeV to tens of TeV, a range which is of great interest for astro-particle physics. The details of this detector are still under development and are awaiting positive flight results from the Proto-type SRD which is scheduled to fly on STS-107 in June, 2001.

5.4 TRANSITION RADIATION DETECTOR (TRD)

5.4.1 Structure

The role of the TRD is to discriminate between e^-/p^- and e^+/p over the range E = 3 – 300 GeV. This is accomplished by detecting the presence of X-ray photons emitted by electrons and positrons when they pass through a radiator. p, p^- and nuclei do not emit such radiation. The radiation is detected in proportional tubes filled with preferentially $Xe:CO_2$ (80:20) [or $Xe:CF_4$ (80:20)]. Xenon is chosen as it gives a very high efficiency for photon detection. By proportional multiplications, the ionization electrons are converted into a measurable signal.

The TRD detector is composed of 5248 proportional tubes which are made from a multi-layer wound composite structure. The composite includes layers of poluyurethane, conductive carbon-polyamide, aluminum and kapton. The straw tubes are grouped into 44 separate segments which are connected through gas manifolds. The straws have an inner diameter of 0.24 inch (6.02 mm), a wall thickness of 3.0e-3 inches (72 µm) (See Figure 5.4.1.1) and varies in length from 59.1 inches to 86.6 inches (1.5 m to 2.2 m).

A straw module consists of 16 straws (See Figure 5.4.1.2, prototype module) glued together with 6 CFC stiffeners (I-Beam 0.01 inch (0.3 mm) thick) running alongside the straws. Every 3.94 inches (10 cm) strips are glued across the module for extra rigidity. The straw ends are glued into polycarbonate endpieces. The endpieces contain the wire fixation pieces (wire: gold plated tungsten, 0.001 inch (30 μm) diameter; wire fixation pieces: Cu/Te alloy), the gas distributor and the gas seal.

The TRD is built from 20 layers of the straw modules where a gap of 0.91 inch (23 mm) between the layers is filled with a radiator material (polypropylene fibers). The upper 4 layers (72 modules) and the lower 4 layers (56 modules) are oriented in the X-direction and the 12 middle layers (200 modules) in the Y-direction (See Figures 5.4.1.3 & 5.4.1.4)).

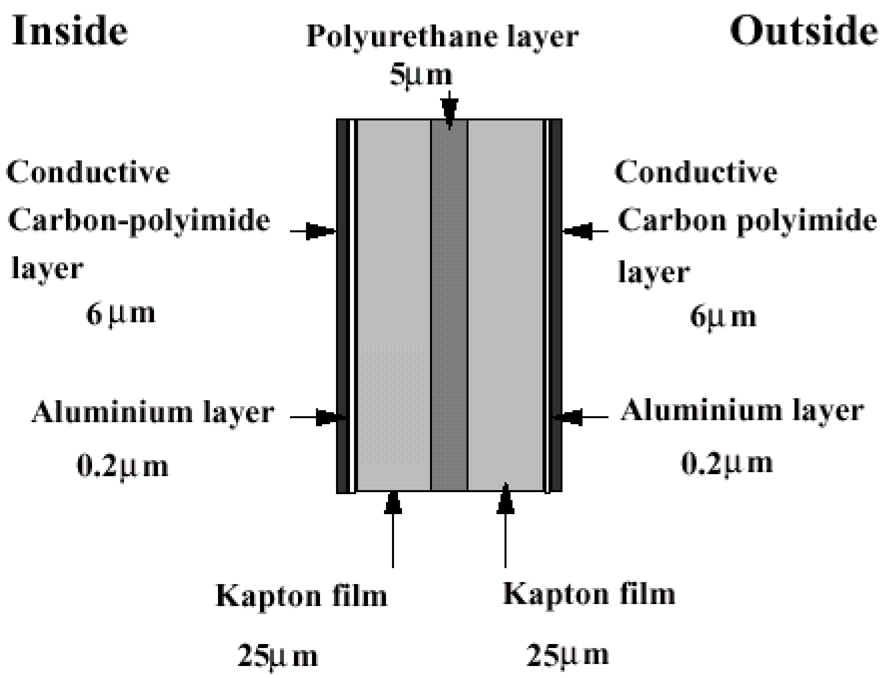


Figure 5.4.1.1 – Composition of Straw Wall

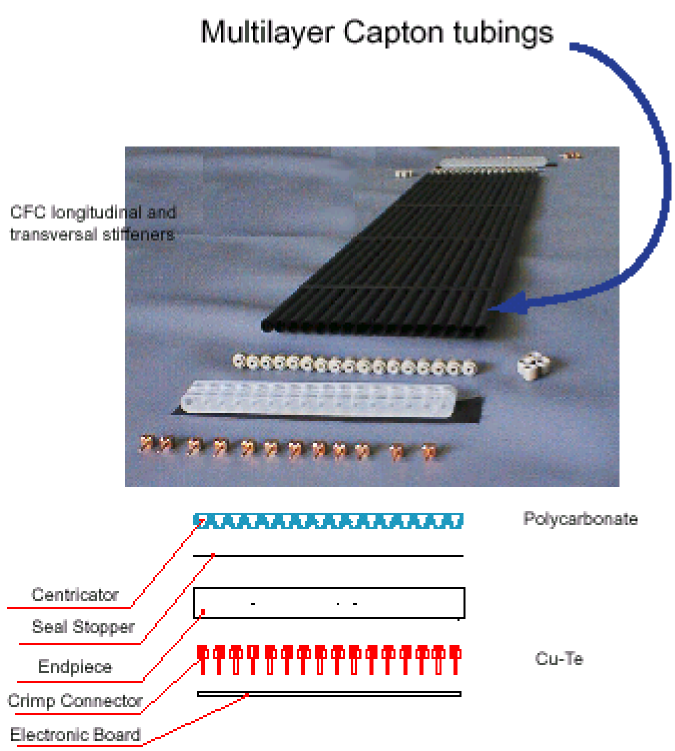


Figure 5.4.1.2 – Straw Module Production

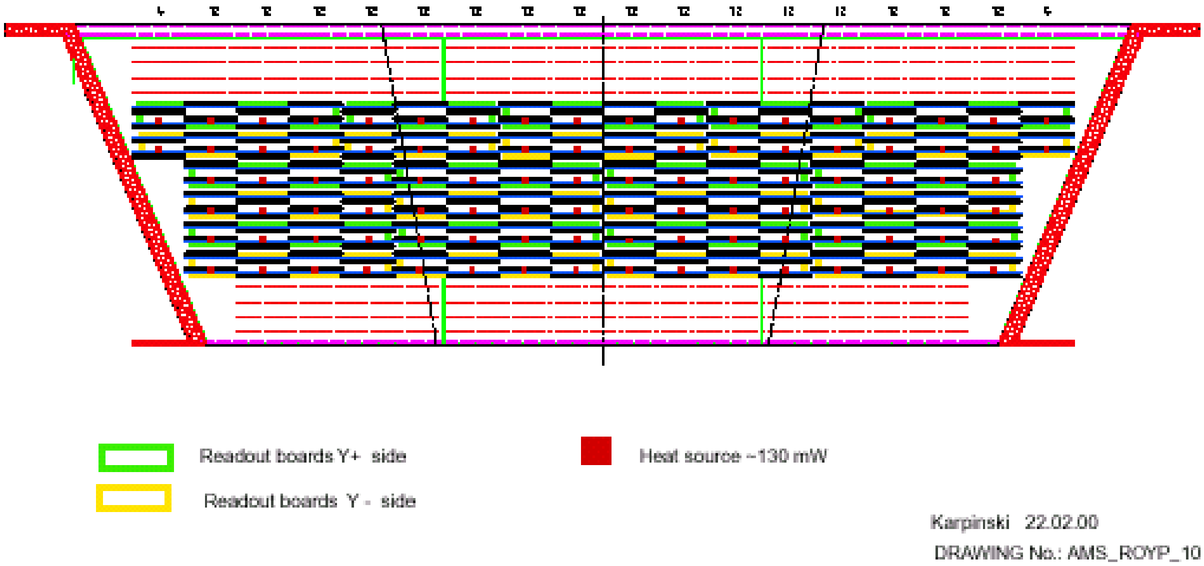


Figure 5.4.1.3 – TRD X-Z Cross Section

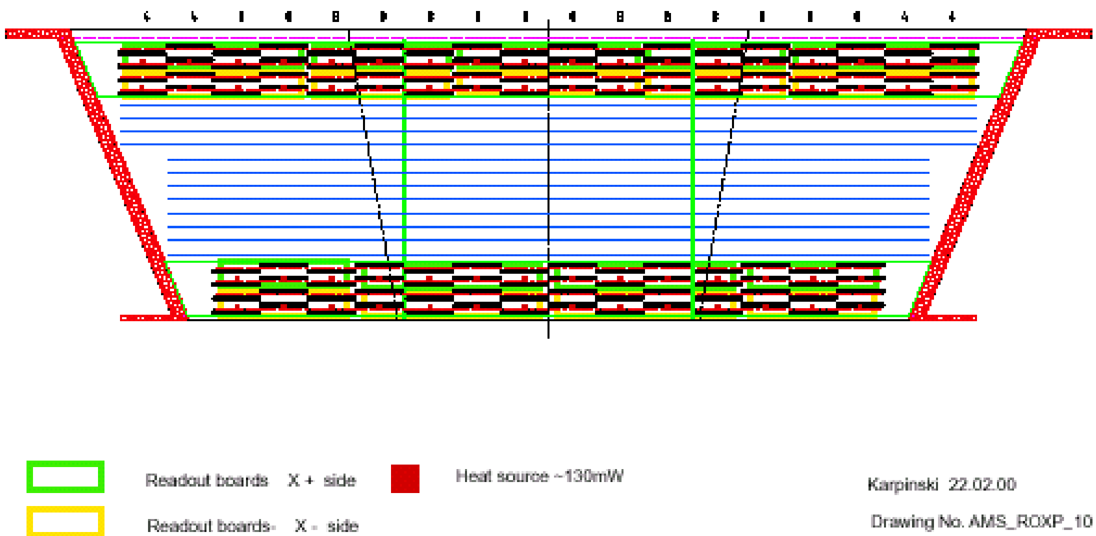


Figure 5.4.1.4 – TRD Y-Z Cross Section

The 20 layers of straw modules and radiators are mounted in an octagon structure which consists of 8 honeycomb (HC) side panels (1.18 inches (30 mm) thickness), a lower HC support plate and an upper HC plate (See Figure 5.4.1.5). The size of the octagon structure is 91 inches x 24.5 inches (height) (2.3 m x 0.6 m), and it weighs ~1000 lbs (454 Kg) (including the weight of the upper TOF HC). Inside the octagon structure, the straw modules are further supported by 4 bulkheads (0.1 inch (3 mm) thick), 2 in the Y-direction and 2 times 2 smaller ones in the X-direction (See Figure 5.4.1.6).

The TRD is located below the SRD and above the upper TOF on the experiment stack. The octagon structure is mounted to the USS-02 at 4 locations, just above the vacuum case interface. It is mounted together with the upper Time of Flight (TOF) system on top of the tracker system on two separated honeycomb support plates.

The frontend readout-electronics and the HV-supply units are mounted on special boards close to the module endpieces (See Figure 5.4.1.7). A dedicated electrostatic shielding of the electronic boards is foreseen. The Xenon gas distribution system is also mounted close to the ends of the modules on the opposite side of the electronics (See Figure 5.4.1.7).

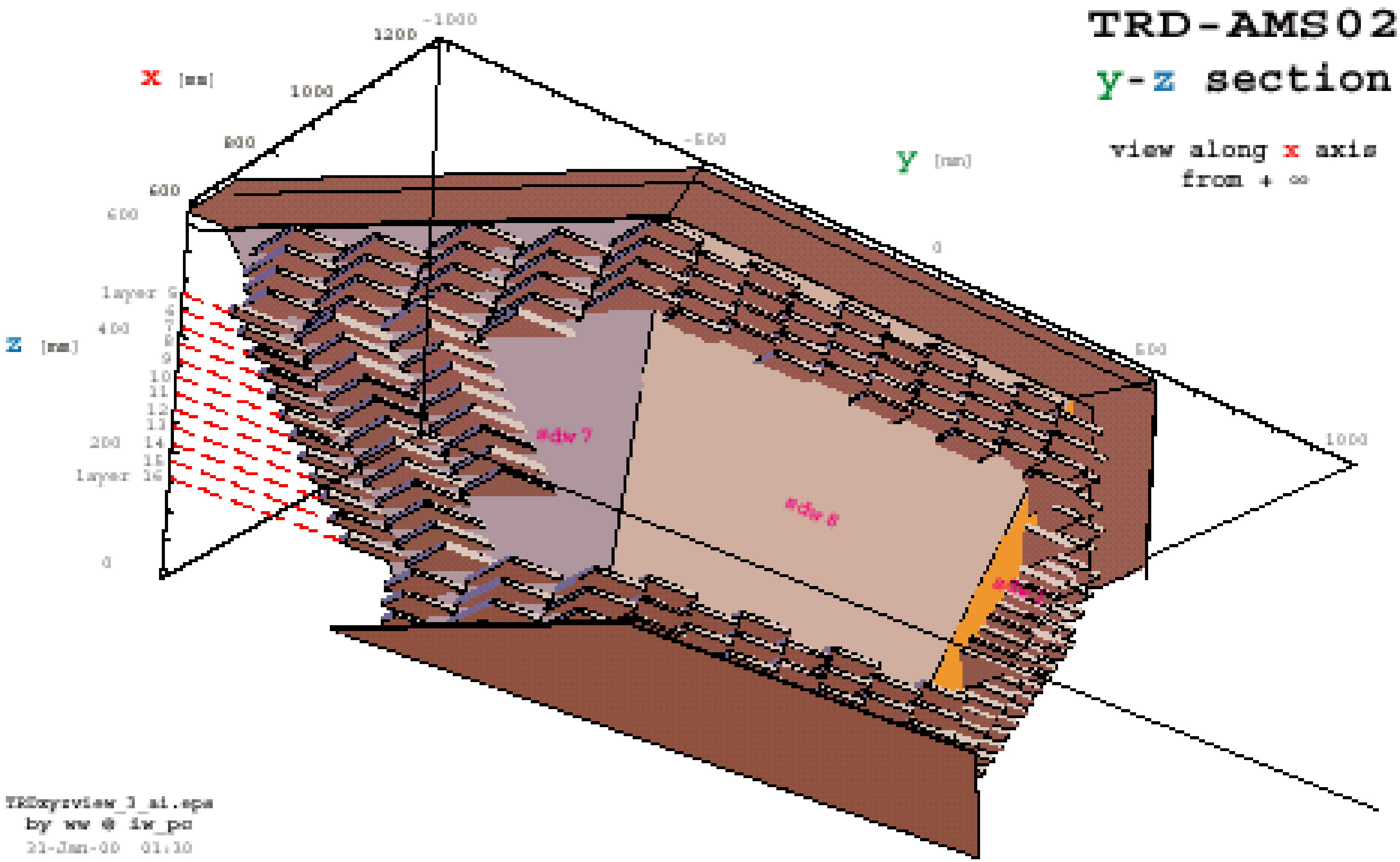


Figure 5.4.1.5 – TRD Octagon with Straw Modules

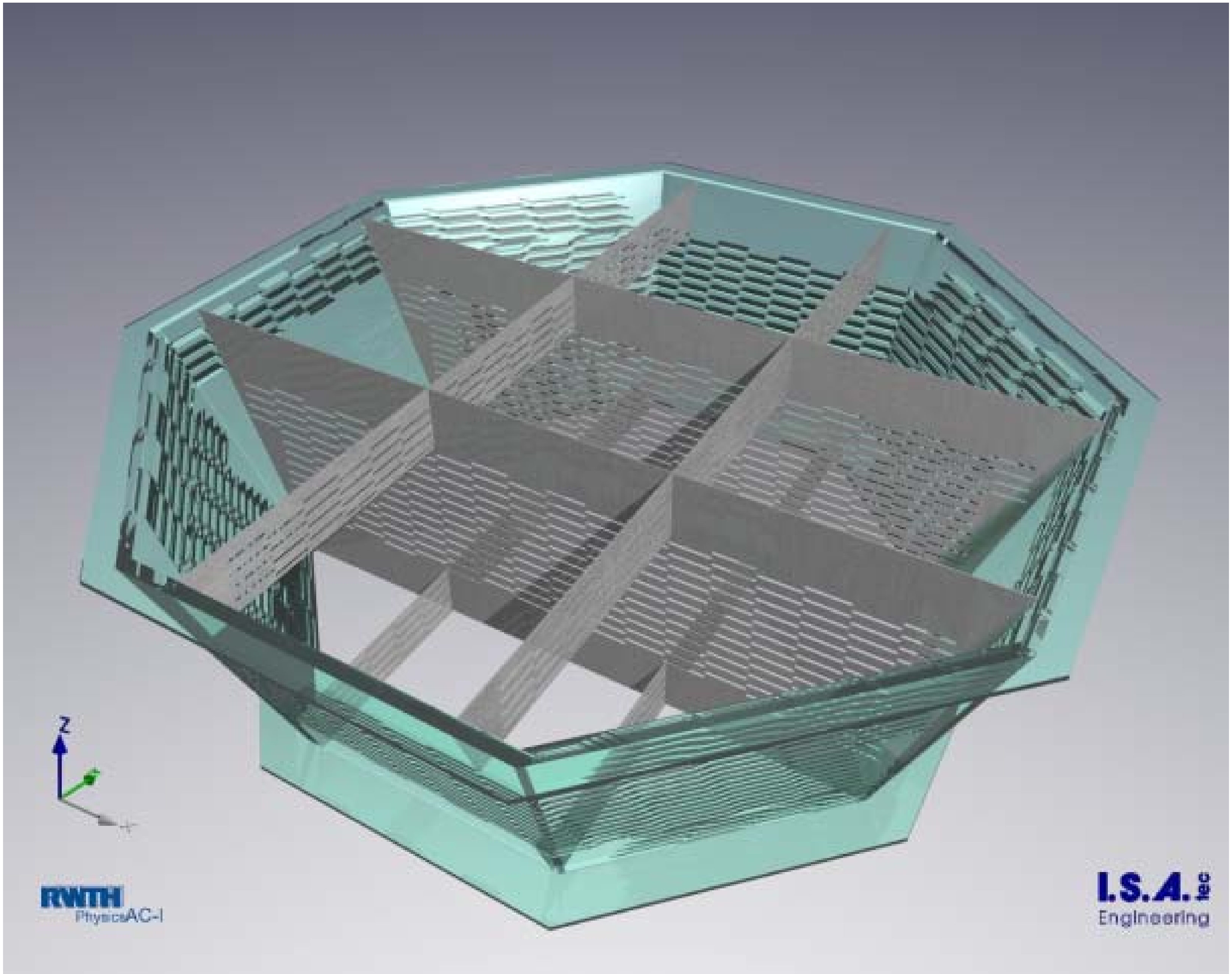


Figure 5.4.1.6 – TRD Bulkheads Inside the Octagon

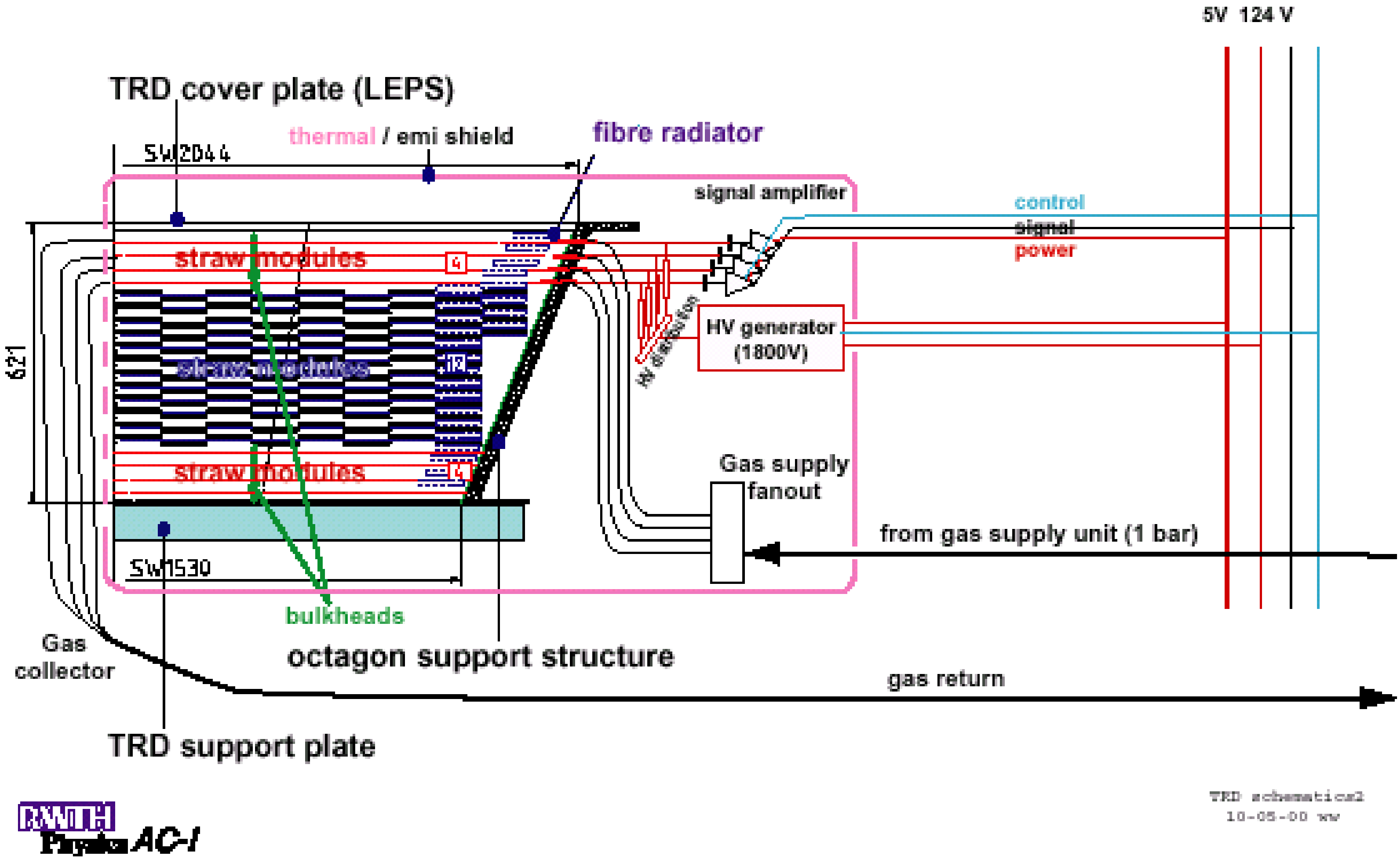


Figure 5.4.1.7 – TRD System Components

5.4.2 TRD Gas Supply System

The TRD Gas Supply System supplies a mixture of 80% Xenon (Xe) and 20% Carbon Dioxide (CO_2) or Freon (CF_4). The density and purity of the gas mixture is monitored and corrected to ensure efficient photon detection. The gas supply system includes three tanks, one for the Xe, one for the CO_2 or CF_4 (TBD), and one mixing tank (See Figures 5.4.2.1, 5.4.2.2 & 5.4.2.3). These tanks are all housed in a box to protect them from meteoroids and orbital debris. The box is mounted to the USS-02 on the wake side (for ISS flight). This location also helps to protect them from damage.

The Xe tank is a composite over-wrapped stainless steel tank that is designed and built by Arde, Inc. This tank is the same design as one that is used on the Plasma Contactor Unit for ISS. It has a maximum design pressure of 3000 psig with a minimum temperature rating of –60°F and a maximum temperature rating of 150°F. The relief valve is set to 2800 psig and the normal operating pressure is 1550 psia. The normal operating temperature is 77°F (25°C). The tank was designed with a proof test factor of 1.5 x MDP and a minimum burst factor of 3.1 x MDP. It has an outside diameter of 15.37 inches (390 mm) and a volume of 1680 cubic inches (27.5 liters). It can carry up to 109 lbs (49 Kg) of Xe and has been tested to 8.9 Grms at 0.08 g^2/Hz.

The CO_2/CF_4 tank is a composite over-wrapped stainless steel tank that is also designed and built by Arde, Inc. This tank was designed for use on the X-33 vehicle and has a maximum design pressure of 3200 psig. This tank operates at 77°F, but has a minimum operating temperature of –100°F and a maximum operating temperature of 300°F. The relief valve is set to 2800 psig, and the normal operating pressure for CF_4 is 1740 psia (much less for CO_2). The tank is designed with a proof test factor of 1.5 x MDP and a minimum burst factor of 2.0 x MDP. The outside diameter is 12.42 inches (315 mm) and it has a volume of 813 cubic inches (13.3 liters). The tank weighs 9.5 lbs (20.9 Kg) and it can hold a maximum of 18 lbs (8.2 Kg) of CF_4 (8.5 lbs (3.9 Kg) CO_2). A vibration test has been performed to 8.9 Grms at 0.07 g^2/Hz axially and 4.5 Grms at 0.02 g^2/Hz laterally.

The small mixing tank will also be manufactured by Arde, Inc. It will have a nominal operating pressure of 60 psia and a normal operating temperature of 77°F. A proof test factor of 1.5 x MDP and a minimum burst factor of > or = 2.0 x MDP will be used. The volume will be ~122 cubic inches (2 liters), but the rest of the parameters are currently TBD.

The fittings and connections in the gas system include stainless steel tubing, welded joints, and numerous gas manifolds. The stainless steel tubing will range from 0.1 – 0.25 inch (3 – 6 mm) outer diameter. Connections will be made with welded joints (as an alternate, metal sealed fitting could be used). The connections between the gas manifolds and the TRD segments are made with 0.04 inch (1 mm) inner diameter Polyether Ether Ketone (PEEK) tubing and metal connectors.

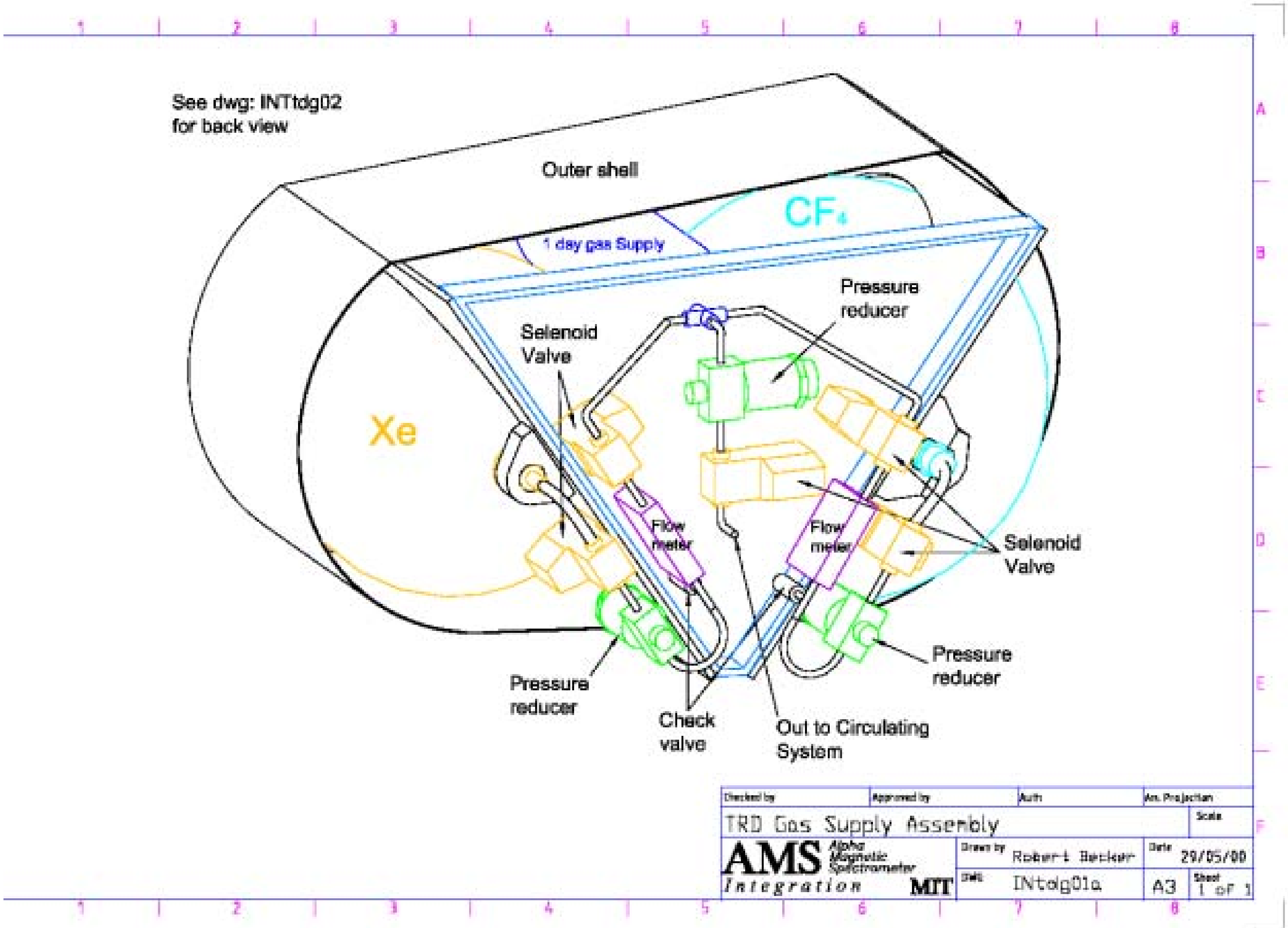


Figure 5.4.2.1 – TRD Gas Supply Assembly (Front View)

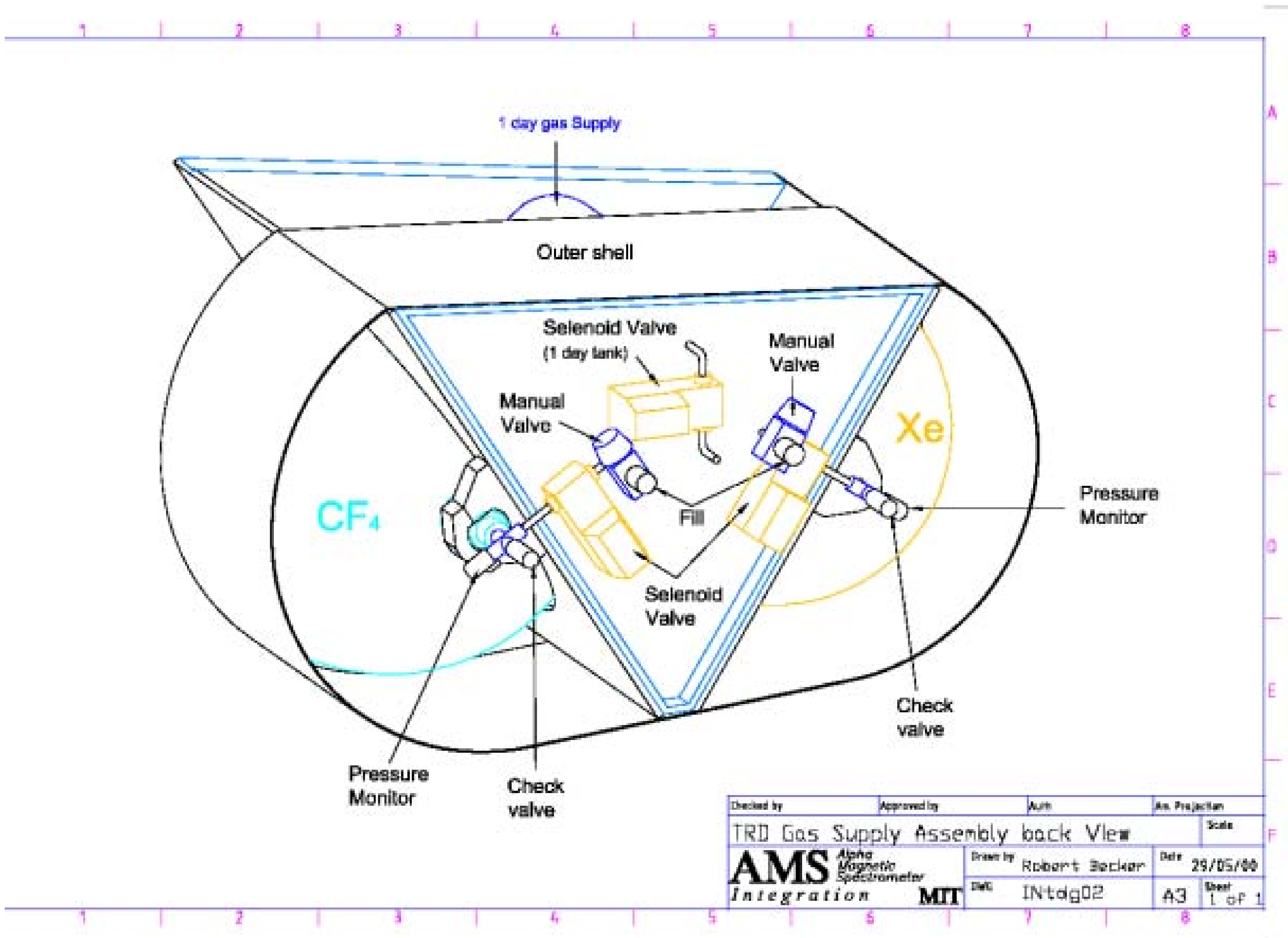


Figure 5.4.2.2 – TRD Gas Supply Assembly (Back View)

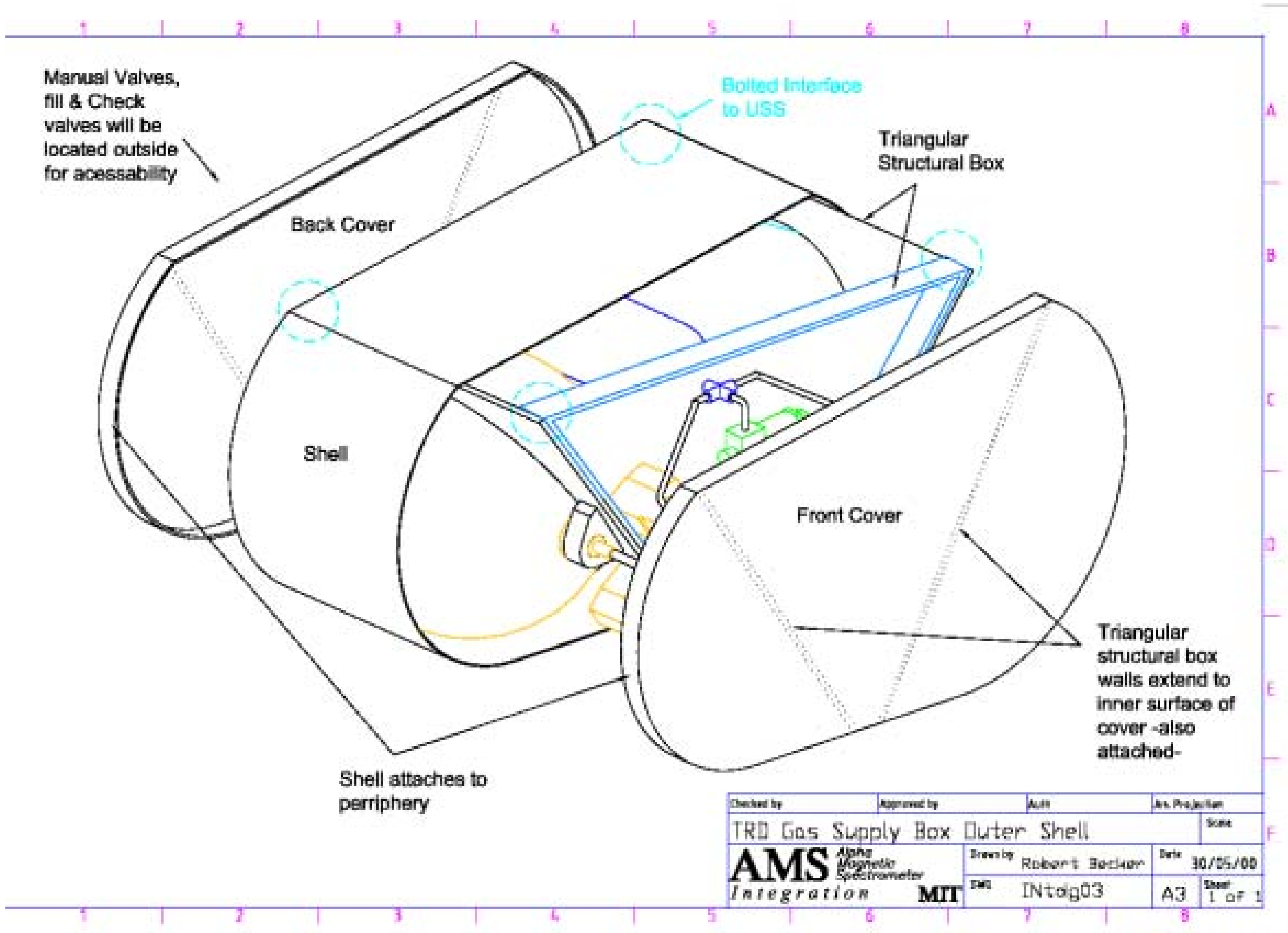


Figure 5.4.2.3 – TRD Gas Supply Assembly With Covers

The TRD straw tubes have a maximum design pressure of 30 psia. The minimum and maximum design temperature is still TBD, but testing is ongoing. The relief valves will be set to 30 psia. The normal operating pressure is 14.7 to 17.5 psia on orbit and 17.5 psia on the ground. The proof test factor of 1.5 x MDP will be employed and a minimum burst factor > or = 2.0 x MDP will be employed. Each of the 44 separate segments contain 490 cu. in. (8 liters) of gas. The nonflammable gas mixture is circulated through these tubes in a continuous loop. The density and purity of the gas mixture is monitored and corrected.

The 41 TRD segments are connected through manifolds to the Box C, containing controls, monitors, and recirculation pumps. Box S contains all gas supplies with a limited mixture volume, a feed control activated by computer and a density monitor. The general layout is shown in Figure 5.4.2.4. All pressures are given at 77^{o}F (25°C). The 41 sealed TRD containers of approximately 490 cu. in. (8 liters) are at 17.4 psi, components in box C are estimated at 150 cu in at < 30 psi and box S is described below.

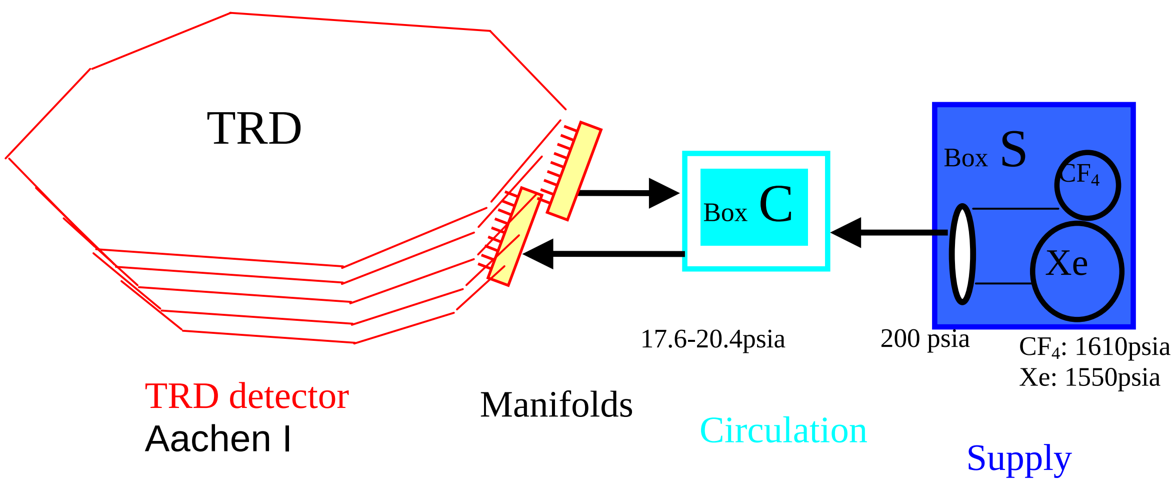


Figure 5.4.2.4. Schematic Arrangement of the AMS-TRD gas system. All pressures are given at 77 F (25 C). Updated

Box S Description

Box S contains the resupply system to ensure up to 7 liters/day of gas may be provided to the TRD system. The schematic of the system is shown in Figure 5.4.2.5.
A components list is also shown.

Box S Operations

Operation: Once per day: By computer control open V1, V2, V3 sequentially, one at a time until desired partial pressure is reached in the daily supply bugger (D). Cross check with redundant pressure sensors P3 and P6. Repeat procedure with CF_4 (CO_2) branch.

Transfer: Several times per day V4 is opened under computer control to release fresh gas from the buffer volume (D) into the control module Box C containing the circulation system. Transfer is limited to < 7 liters at 1 atm/day in normal mode.

Normal Operation:

- During launch and flight normally all valves of Box S are closed.
- P1a,b, P3ab and temperature probes are monitored in 5-10 minute intervals.
- Once per day on special command the computer initiates fillup of the daily bugger D from rest gas pressure of 20 psia to 200 psia.
    - A. Xenon transfer into vessel D
        1. Open valve V1a, filing the volume of the connecting valves (~100 cm3) to V2a.
        2. Close V1a, wait ~10 seconds. Open V2a so the limited volume discharges through the flow restrictor into vessel D. Verify by P3a,b increase.
        3. Repeat step 1a,b) until P3ab reaches ~164 psia. Verify by P3a,b. All connecting lines between solenoids are at 20-30 psia now.
    - B. CF4 transfer
        1. Repeat the same sequence of operation 1,2 with V1b, v2b, V3b. Note volume between V1b and V2b is 20 cm3 only.
        2. Add 34 spia of CF4 so vessl D reaches ~200 psia.
        3. Close all valves. Verify all pressures to remain constant.

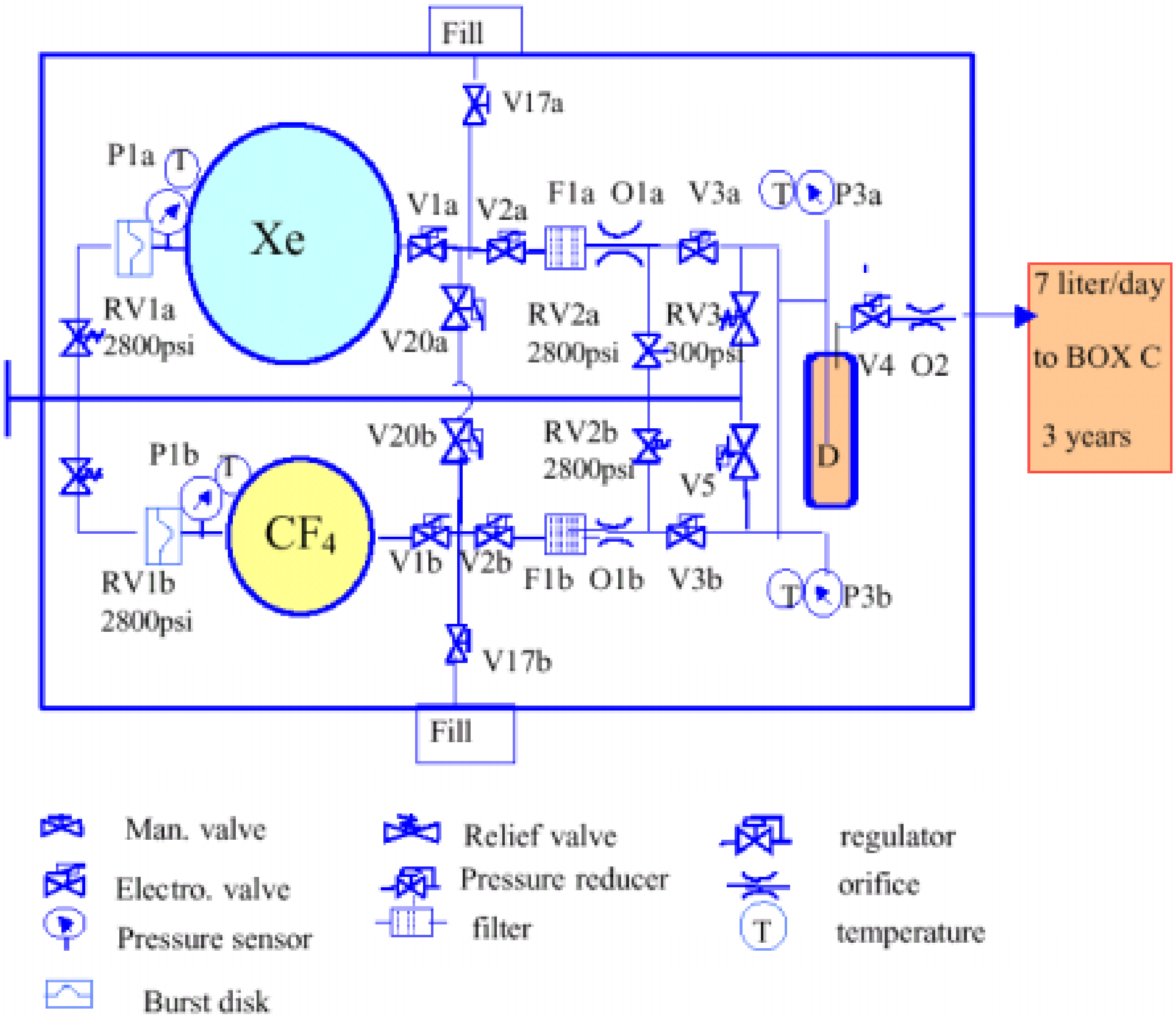
  After this step the daily ration of 80:20 volume percent mixture is provided in D. All solenoids are closed, connecting lines at low pressure. The next step can wait.

    - C. Transfer to TRD
        1. Open V4 and the corresponding valve V6 in Box C for 30 seconds to release gas into the circulating gas system of the TRD through the flow restirctor O2.
        2. Verify transferred volume by P3a,b drop.
        3. Repeat for fine tuning.

Safety Issues

- Solenoids are always closed, unless current flows. 24 V supply is interlocked.
- Relief valve Valcor V5-500-77-3: crack pressure 2800$\pm$150 psig, open flow rate: 25 SCFM (min), Worst case fire with 1 degree C/min increase releases 1.9 SCFM/min << 25 SCFM for Xenon. For CF4 maximum release at 1 degree C/min. (0.4 SCFM)
- Flow restrictors cannot blow out, since they are in a recessed container.
- All components operable –20 to +65 degrees C. No gas flow or pressure build up since solenoids are closed.

Note: The maximum possible transfer of 7 liters causes in the total TRD volume a maximum pressure increase of 2%, whereas a 50% is design operation limit.



pressures, normal	**Xe (25^{0}C) 1550 psia**	**CF_4(25^{0}C) 1610 psia**
Vessel rated at:	**Xe (65^{0}C) 3000 psia**	**CF_4(65^{0}C) 3200 psia**

Figure 5.4.2.5 - Box - S   TRD Gas Supply (Updated)

Components		Weight (kg)
Xenon		49.5
CF_4		8.2
Xe Vessel	ARDE D4636, 1680 in^3, Same as ISS PCU	8.6
CF_4 Vessel	ARDE D4683, 813in^3, Same as X33	4.3
D - vessel	Arde SDK 13181, 61 in^3, Same as ISS SAFER	1.0
Solenoid valves V1a,b, V2a,b, V3a,b, V20a,b, V4, V5		
Marotta MV 100	(current opens) 11x .37	4.1
Pressure monitors P1a,b, P3a,b, Box C 3		
GP50 model 7900 including T-measurement	9x.04	0.4
Self closing valves., V17a,b CERN Type		0.1
Relief valves, RV1a,b ,RV2a,b, RV3 Valcor P/N V3500-77		

2750±5% psia 3x.191g	0.7
O1a,b ,O2 flow restrictors. approx. 50l/h	
Lee Visco jet VDCA183041H 4x	0.3
Manual valves, V17a,b	*0.8*
Filters, F1a,b	*0.1*
Support box	*20*
Piping	*2*
Total Box S	*103.4*

Box C Description

The schematic for Box C is shown in Figure 5.4.2.6. The circulation pumps serve to move the gas through the system to ensure a uniform environment. The purifier will be mounted on the outside of Box C ground operations and will be removed prior to flight. It is connected by spring loaded valves which will be capped before launch.

Box C Operations

Normal – gas circulates through the straws, no purification.
Gas cleaning – on earth to remove oxygen and water from gas.
Daily filling – replace gas lost in normal operation.
System pressure test – check for loss anywhere in straw system by looking for pressure change when closed.
Segment pressure test – isolate single segment and monitor pressure while isolated.
System filling – initial loading of Xe into storage/recovery system.
Recovery Xe from AMS-02 – remove Xe from AMS-02 Xe storage tank in Box S.
AMS-02 filling – transfer Xe from storage/recovery vessel to AMS-02
Gas composition check – check for contamination of gas with residual gas analyzer.

Operations Before Launch

1. System should be filled and circulating through the purifier well before launch. Circulation should continue until as close to launch time as possible.
2. Before power off for launch, all segment valves should be open. Box C and Box S valves should be set as specified.
3. Purifier removed, spring loaded valves capped.

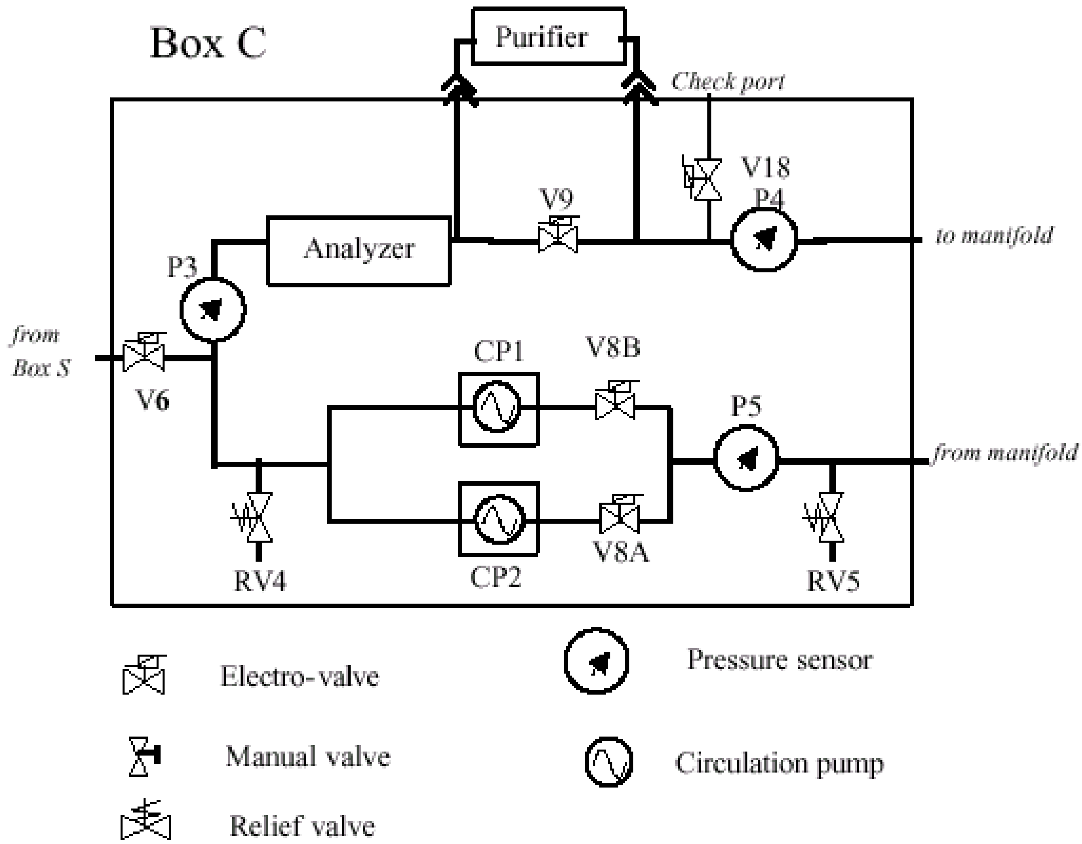
Operations after Launch at Startup

1. Isolate all segments
2. Check PNNa-b (see figure 5.4.2.7) for losses in each segment.
3. Open all working segments.
4. Start CP1, begin normal operation.

Operations for Return

1. Stop CP1/2.
2. Close VNNa-d (see figure 5.4.2.7) to seal each segment.

Goto landing mode.



Note: The purifier is used to remove O_2 during ground operations and will be removed prior to flight. The attachment points will be covered and sealed.

Figure 5.4.2.6 - Box C - TRD Gas Recirculation System (updated)

Box C Components List

1. Circulation pumps Cp1,2 – KNF-Neuberger NMP 30 KNDC iron free, brushless membrane pump.
2. Valves V8A,B, V9 – Burkert Type 6123 micro flipper solenoid valves. Isolating membrane.
3. Valves V6, V18 – Marotta direct acting solenoid valve, MV 100
4. Pressure Sensors P3, P4, P5 – either three pressure/temperature sensors from GP:50 series 7900 (150 psia maximum) or Honeywell 24 PC series differential sensors
5. CO2 sensor – Square One technology Sidestream Gas Analyzer.
6. Purifier – Oxisorb
7. Relief valves RV4, RV5 – Straval RV A05, set at 25 psia.

<u>Straw Tube Segments</u>

From the Box C assembly, 6mm stainless steel gas lines run to the top rim of the TRD, where input and output manifolds are located. The 5248 tubes of the TRD are grouped into 41 separate segments, each separately attached to the input and output manifolds

(See Figure 5.4.2.7). Each segment is small enough so as not to be considered a pressure vessel (1 bar×8 liters=0.8 kJ). Each manifold is connected to the 44 TRD segments via pressure controlled isolation valves. 0.1 inch (3mm) steel tubing runs from the isolation valves to the segment inputs and outputs, where it is joined to PEEK tubing (See Figure 5.4.2.8). Where other connections need to be made, Cajon VCR fittings are used.

The isolation system works in two modes. In case of a sudden pressure drop, the control computer will shut all four valves automatically to prevent further gas loss. In case of an increase in gas consumption, or as a periodic check, the computer will close all four valves and monitor the pressure. This will be used to detect slow leaks.
(Failure of any of the shutoff valves or pressure sensors cannot cause MDP to be exceeded.)
(The shutoff valve/pressure sensor assembly will be potted inside a magnetic shielding box to preclude any leak from the gas system volume.)

Backup Information:

The isolation valves will be Burkert Type 6123 2/2 Way Flipper Valves. Closed, they hold 43.6 psi (3 bar) in either direction and have been leak tested to better than 0.1 ml/day loss, 14.5 psi (1 bar) to vacuum through a closed valve. They can be flipped from open to closed and vice-versa by a 12V, 100ms pulse, and otherwise consume no power. They are located near the top flange of the TRD in a region of low magnetic field. The pressure sensors are Honeywell type 24PC.

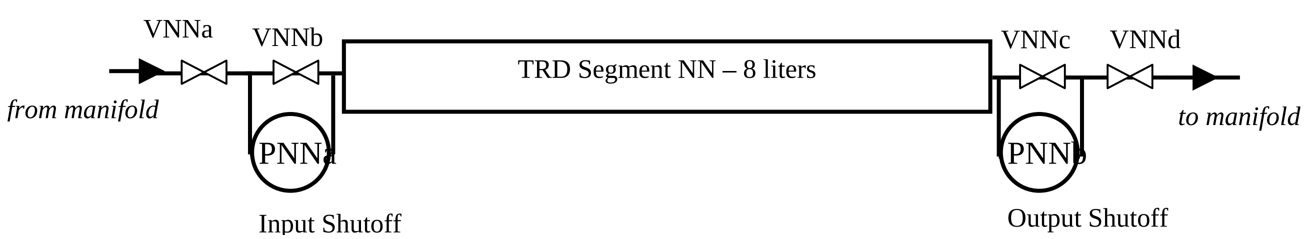


Figure 5.4.2.7 - One of 44 TRD Straw Tube Segments.

High Voltage (HV), Monitoring, and Control

High Voltage (HV), Monitoring, and Control systems consist of a controlling computer, which can execute commands, like close or open the insulation valves in emergency, provide housekeeping data, store calibration constants and adjust the HV to regulate the gas gain. It will condition and perform analog to digital conversion for >100 pressure sensors and >64 temperature sensors distributed around the TRD and the gas system. It also controls the two circulation pumps and provides logic control of >200 gas valves.

The HV system consists of 88 HV boards distributed around the TRD in the vicinity of the readout cards. The control computer, and HV boards are doubled to provide single fault tolerance. The schematic of the HV system is shown in Figures 5.4.2.9 & 5.4.2.10. Each unit provides +1600V (control range: 700-1750V) with current limited to <100μA.

The boards are directly located at the ends of the tubes to which they supply HV, so that there is no exposed HV or cabling.

Mounted inside the Gas Supply System Box C are 2-4 calibration tubes (See Figure 5.4.2.11), which monitor the gas gain changes for locally different temperatures. The calibration tubes have an ID of 0.24 inch (6 mm) like the straw tubes, however are mounted inside a stainless steel container (See Figure 5.4.2.12). On the inner wall is a 0.2μCi deposit of Fe^{55}. The 0.04 inch (1 mm) wall attenuates the 5.9keV radiation to a level less than detectable. The outer stainless steel container seals in the radiation again and supplies the gas for calibration.

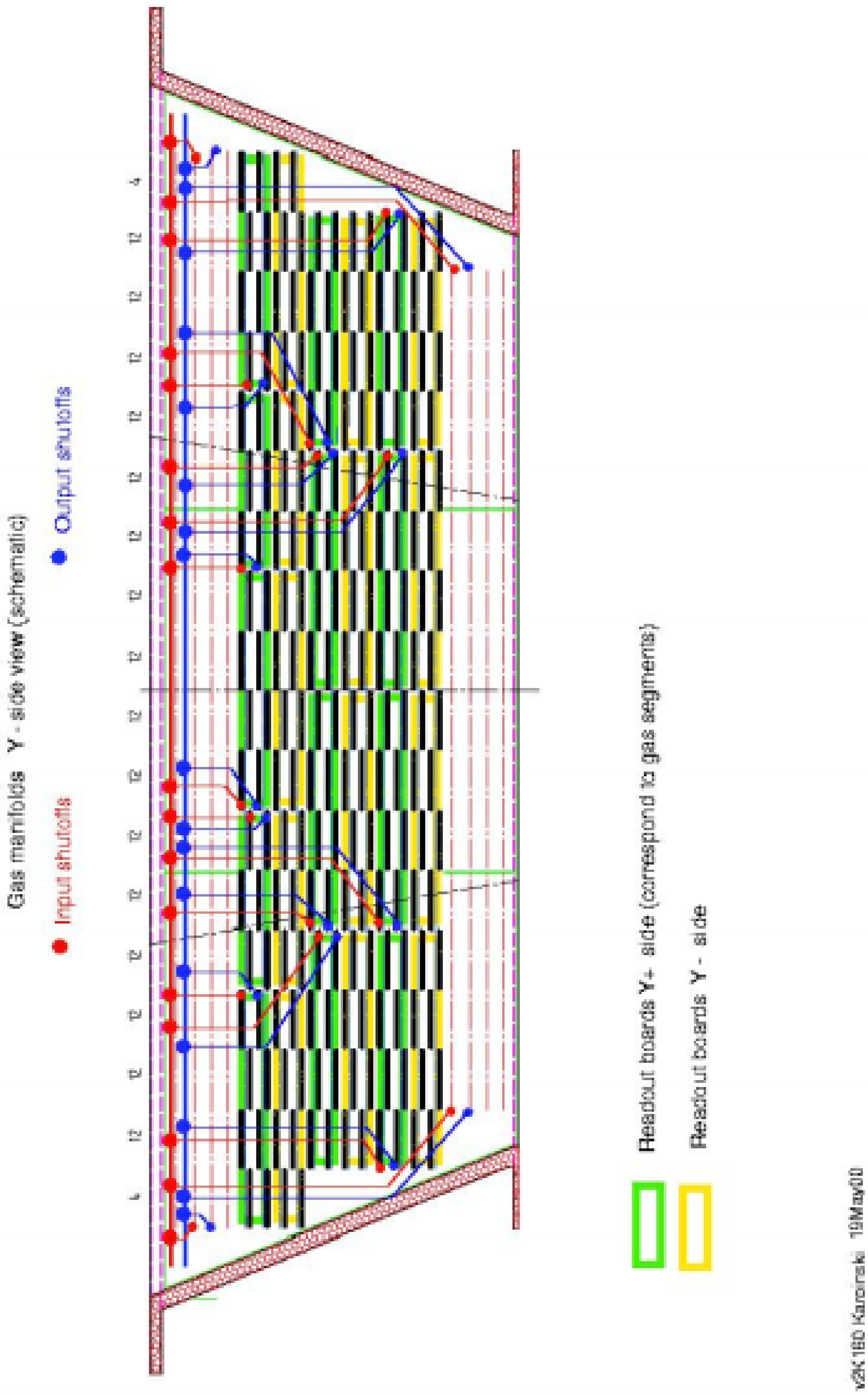


Figure 5.4.2.8 - Gas Manifold Connections to TRD Segments.

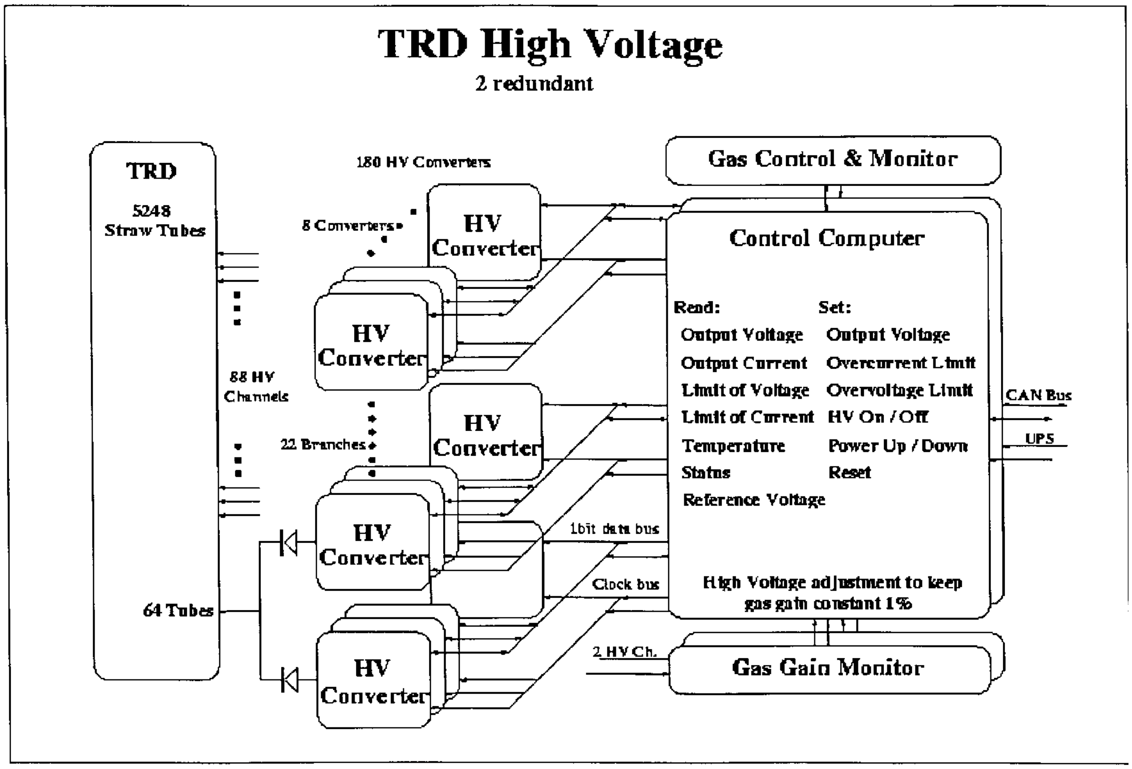


Figure 5.4.2.9 – TRD High Voltage System

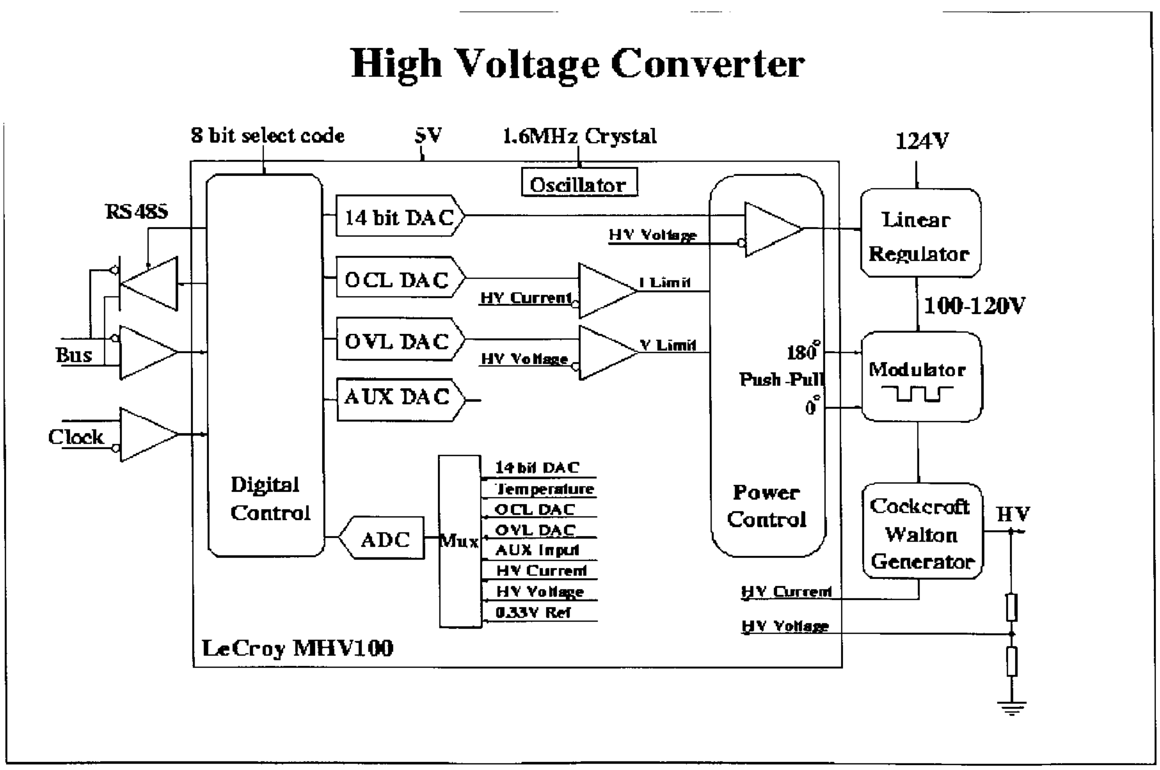


Figure 5.4.2.10 – High Voltage Converter

Calibration Tubes Located in Box C of Gas Supply System – Drawing is TBD

Figure 5.4.2.11 – Location of the Calibration Tubes on the TRD Gas Supply Box C

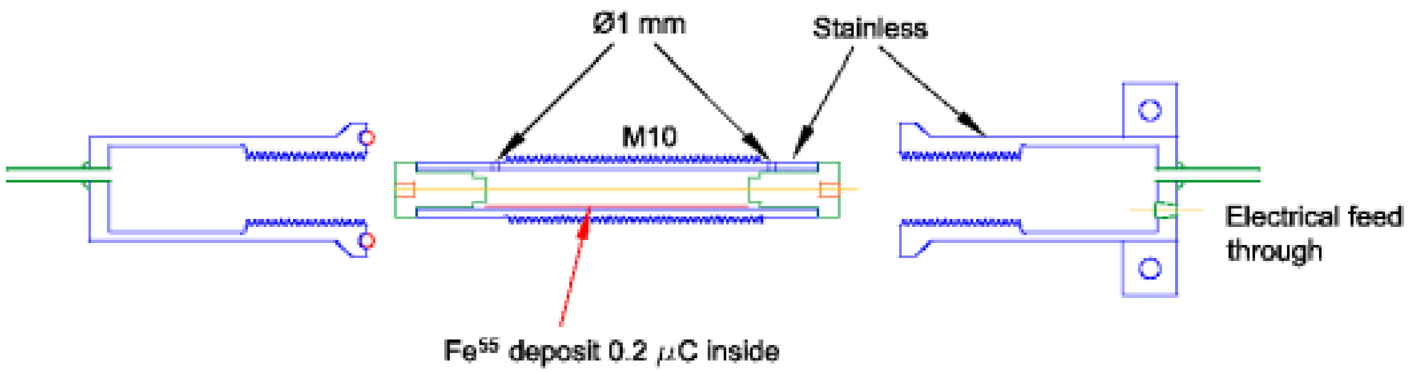


Figure 5.4.2.12 – Calibration Tube with Doubly Contained Weak Source

5.5 TIME-OF-FLIGHT (TOF) SCINTILLATOR COUNTERS

Across the top and bottom of the cylindrical magnet are 4 layers (2 on the top and 2 on the bottom) of TOF scintillator counters. The scintillators provide the trigger function for selection of a single particle or nucleus cleanly traversing the magnet bore. The counters are 10 mm thick and are made of polyvinyl toluene (a Plexiglass-like material) (See Figures 5.5.1 thru 5.5.3). They are enclosed in a cover made of carbon fiber with an aluminum foil surface on the inside and outside. At the ends of each panel are light guides which direct the light of scintillation to photo multipliers. Two large flat aluminum honeycomb panels are used to support the scintillator counters. The upper TOF honeycomb is attached to the Transition Radiation Detector, which is then attached to the USS-02 just above the USS-02-to-Vacuum Case interface. The lower TOF honeycomb is supported to the lower USS-02. The honeycomb panels are roughly circular with a 60.6 inches (1540 mm) equivalent outside diameter. The thickness of the honeycomb aluminum core is 1.97 inches (50 mm) and the aluminum skin is 0.04 inch (1 mm) thick.

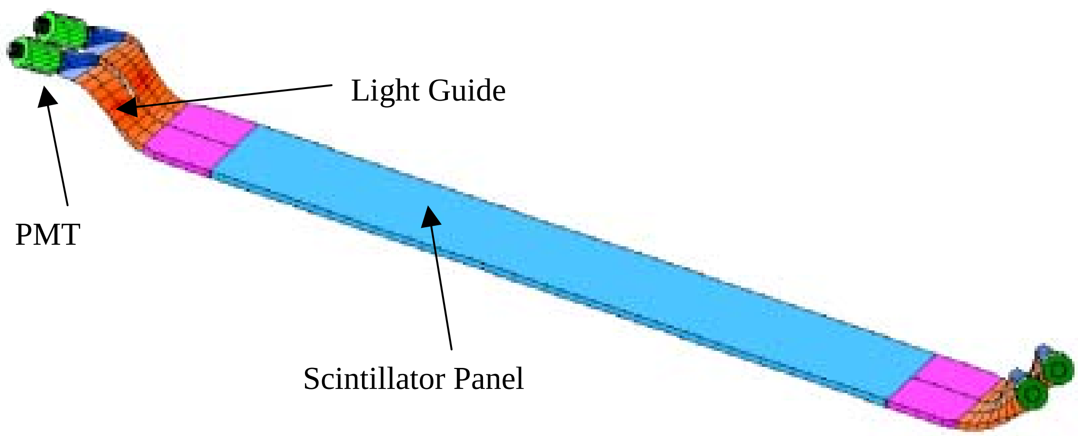


Figure 5.5.1 – Single TOF Counter

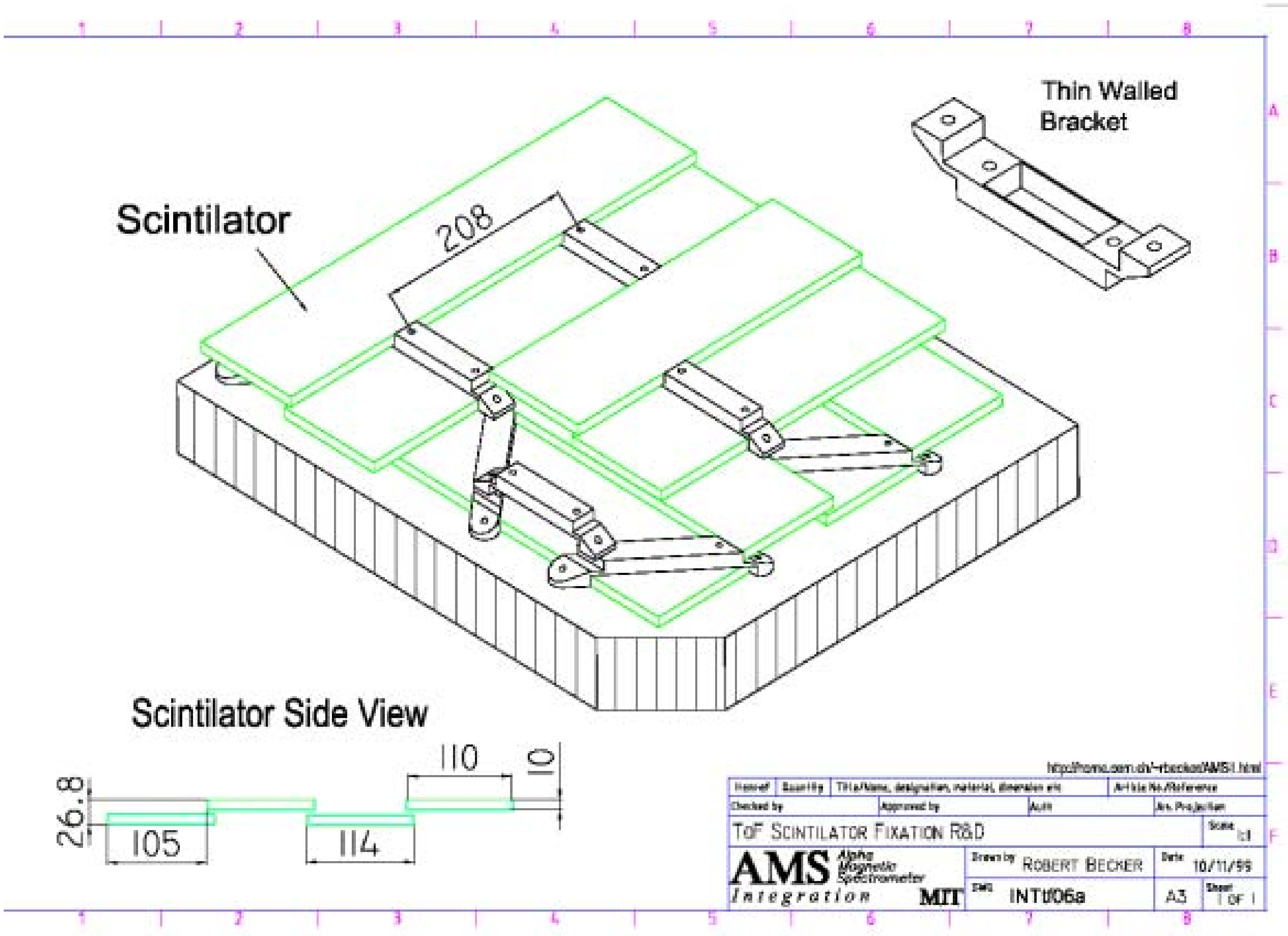


Figure 5.5.2 – TOF Scintillator Fixation

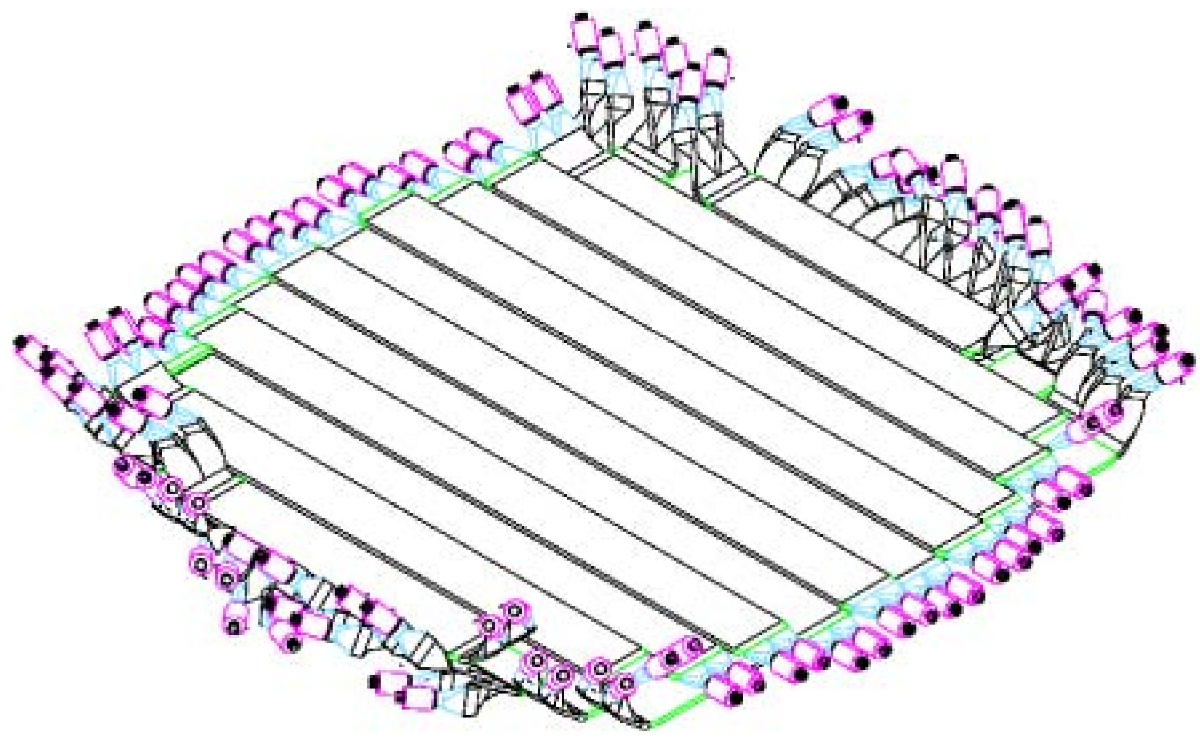


Figure 5.5.3 – TOF Counters Assembly

5.6 RING IMAGING CHERENKOV COUNTER (RICH)

The RICH is located near the bottom of the experiment stack and weighs ~390 lbs (177 Kg). The RICH assembly is shown in Figures 5.6.1 & 5.6.2. The RICH is composed of two primary pieces. The first section contains all of the Photomultiplier Tubes (PMTs) and the second section is the reflector. The first section is made of aluminum cross braces that attach to the USS-02 at 8 locations. The 8 connections to the USS-02 are made with pins/bolts and isolated with Bellville washers. This allows the RICH to be supported without carrying a significant load from one side of the USS-02 through the RICH to the other side. In between the welded cross bracing, the rectangular and triangular PMT units are screwed in place. The PMT units house the numerous PMTs. The reflector section is extremely light (~10-20 lbs) and is made of an aluminum core material with several layers of spray-on gold, alumina, chromium, and quartz. The alumina/quartz is used as the reflective surface. This surface is completely inside the detector and must be light tight. It is likely that a meteoroid and orbital debris shielding will be used on the outside of the reflector, but the details are still TBD. The RICH utilizes an aerogel radiator material, but this material is physically attached to the lower TOF honeycomb.

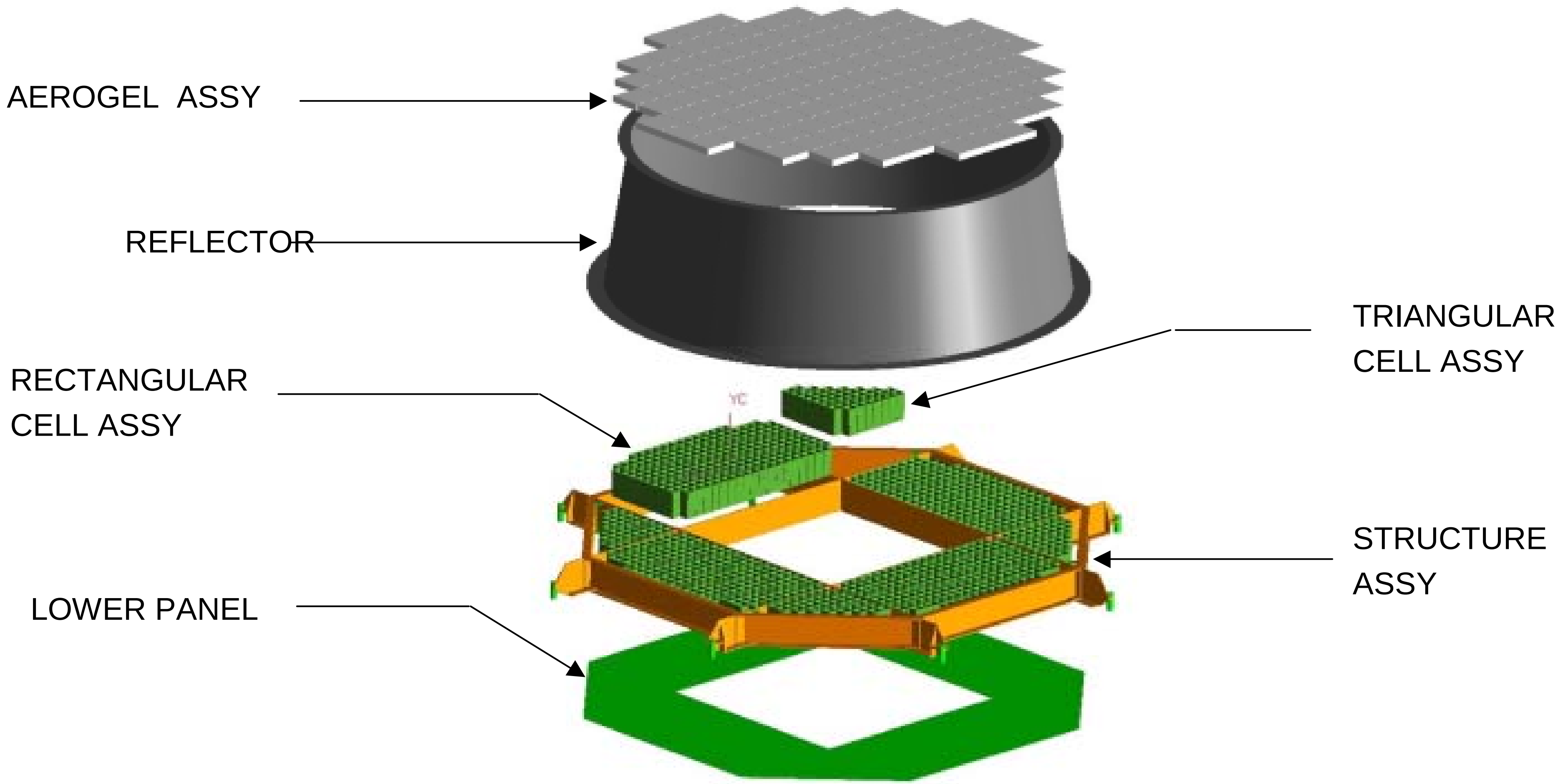


Figure 5.6.1 – RICH Assembly (Exploded View)

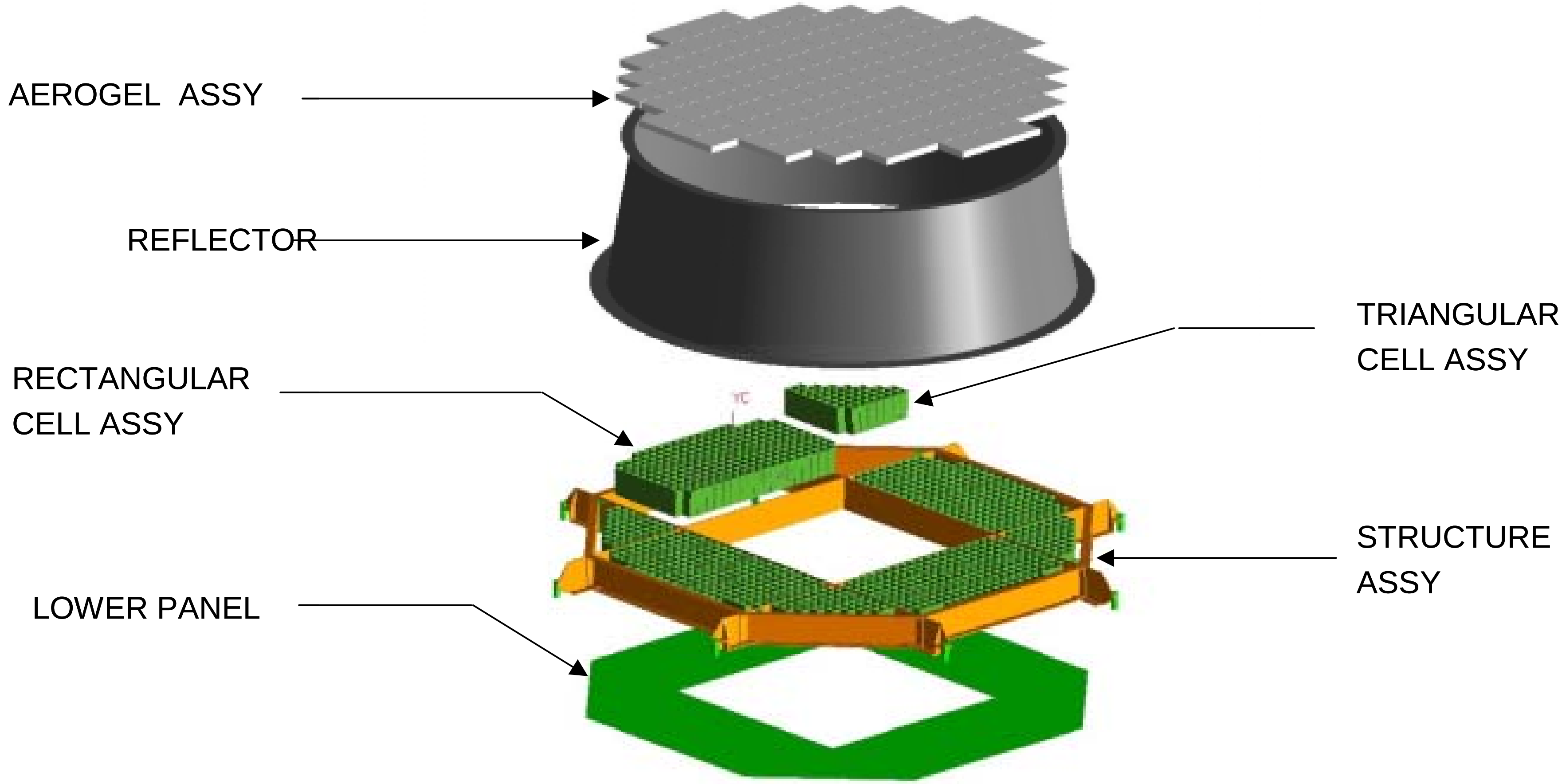


Figure 5.6.2 – RICH Assembly (Lower View)

5.7 ELECTROMAGNETIC CALORIMETER (ECAL)

The main physics goals of the AMS-02 Electromagnetic Calorimeter (ECAL) are: a) to measure the energy of electrons, positrons and gammas up to 1 TeV, b) to ensure the identification of electromagnetic (e.m.) and hadronic cascades with a discrimination capability better than 10^{-4}. To reach these objectives, the ECAL must be able to reconstruct the electromagnetic showers development with high accuracy.

The active part of the ECAL consists of a pile-up of lead foils and scintillating fibers (See Figure 5.7.1). Each lead foil is 0.04 inch (1 mm) thick and is shaped to accommodate scintillating fibers. Small semicircular grooves are machined on both sides of the lead foil. The scintillating fibers are placed in each of the grooves and then another layer of the machined lead is glued to the top and bottom of the first layer. The fibers are glued to the lead foils by means of a bicomponent epoxy glue. The lead foils are squares with 25.9 inch (658 mm) long sides. The scintillating fibers have a 0.04 inch (1 mm) diameter and are 25.9 inch (658 mm) long. Horizontally the fibers are 0.05 inch (1.35 mm) apart, while vertically they are 0.07 inch (1.88 mm) apart.

The active part is subdivided into 9 sub-samples, called "superlayers". Each superlayer consists of eleven layers of the glued lead foils, as described above, with the fibers running in the same direction. Each superlayer is 0.7 inch (18.5 mm) thick. The superlayers are assembled so as to have fibers running in orthogonal directions (X and Y), alternatively. With this kind of structure, e.m. showers development can be studied with the required accuracy.

The ECAL is ~ 31.5 inches (800 mm) square x 9.8 inches (250 mm) high and weighs ~1300 lbs (590 Kg). Nearly half of this weight is due to the lead foils. The mechanical supporting structure for the ECAL comprises a box embedding the sensitive part of the detector and four brackets attaching the box by its four corners to the USS-02 (See Figures 5.7.2 thru 5.7.4). The box is made of 6 elements. The top and bottom pieces are Aluminum honeycomb plates framed with Aluminum. The plates are bolted to four lateral panels along the edges. The four lateral panels are made of Aluminum plate, 15.75 inch (4 cm) thick, carved with squared holes of 1.26 inch (32 mm) sides to house the light collection system. Two sides, serving 4 superlayers, have 72 holes while the two other faces, serving 5 superlayers, have 90 holes each. For each hole, the light collection system consists mainly of a mu-metal square tube for magnetic field shielding, light guides and Photomultipliers (PMs) (See Figures 5.7.5). They are located together with the PM base and front end electronics. An Aluminum backplate is fixed on the rear side of each lateral panel to keep all the light collection systems in the right position and to prevent any displacements of the systems themselves.

Four corner brackets, made of Aluminum plate, link the four plates together and connect the detector to the USS-02 at the bottom of the AMS-02 instrument. The four mounting locations include a radially slotted hole so that the loads of the ECAL are transferred to the USS-02, but the loads from the USS-02 that are transferred into the ECAL are limited.

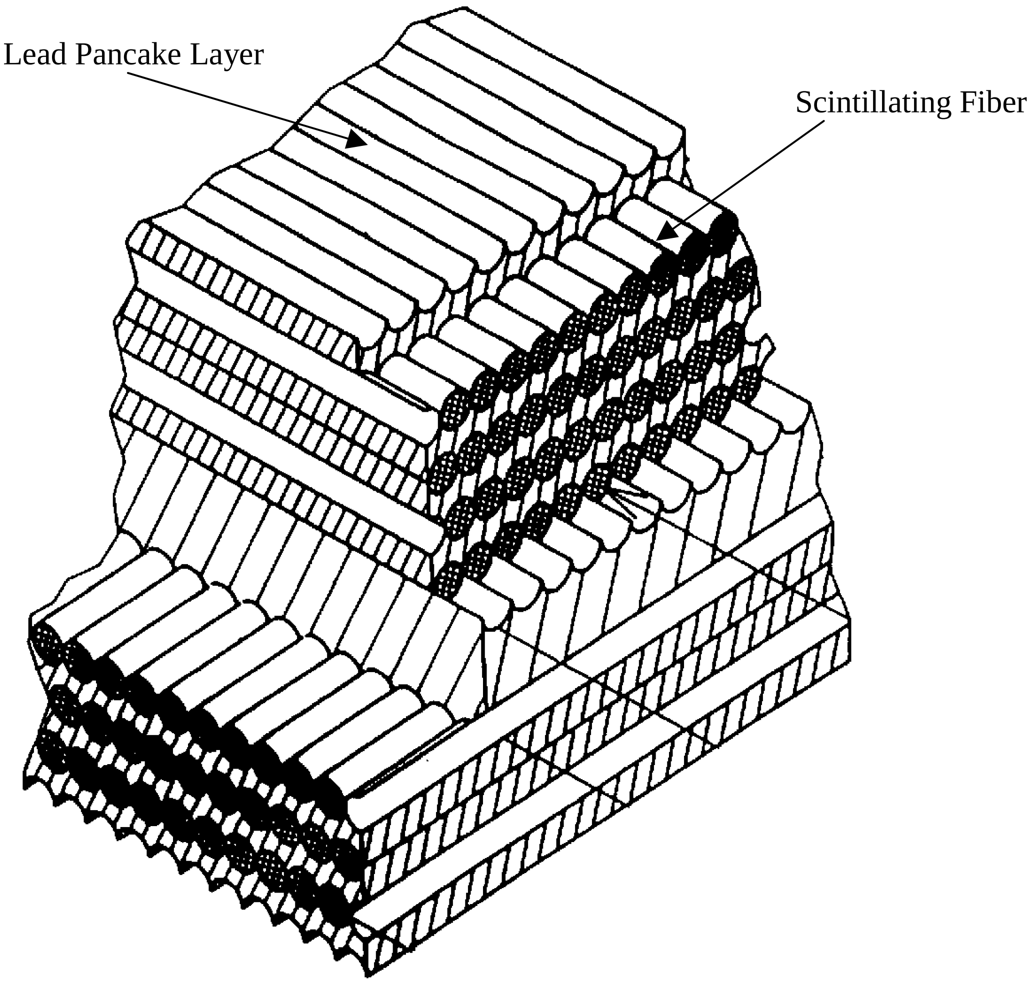


Figure 5.7.1 – ECAL Active Detector

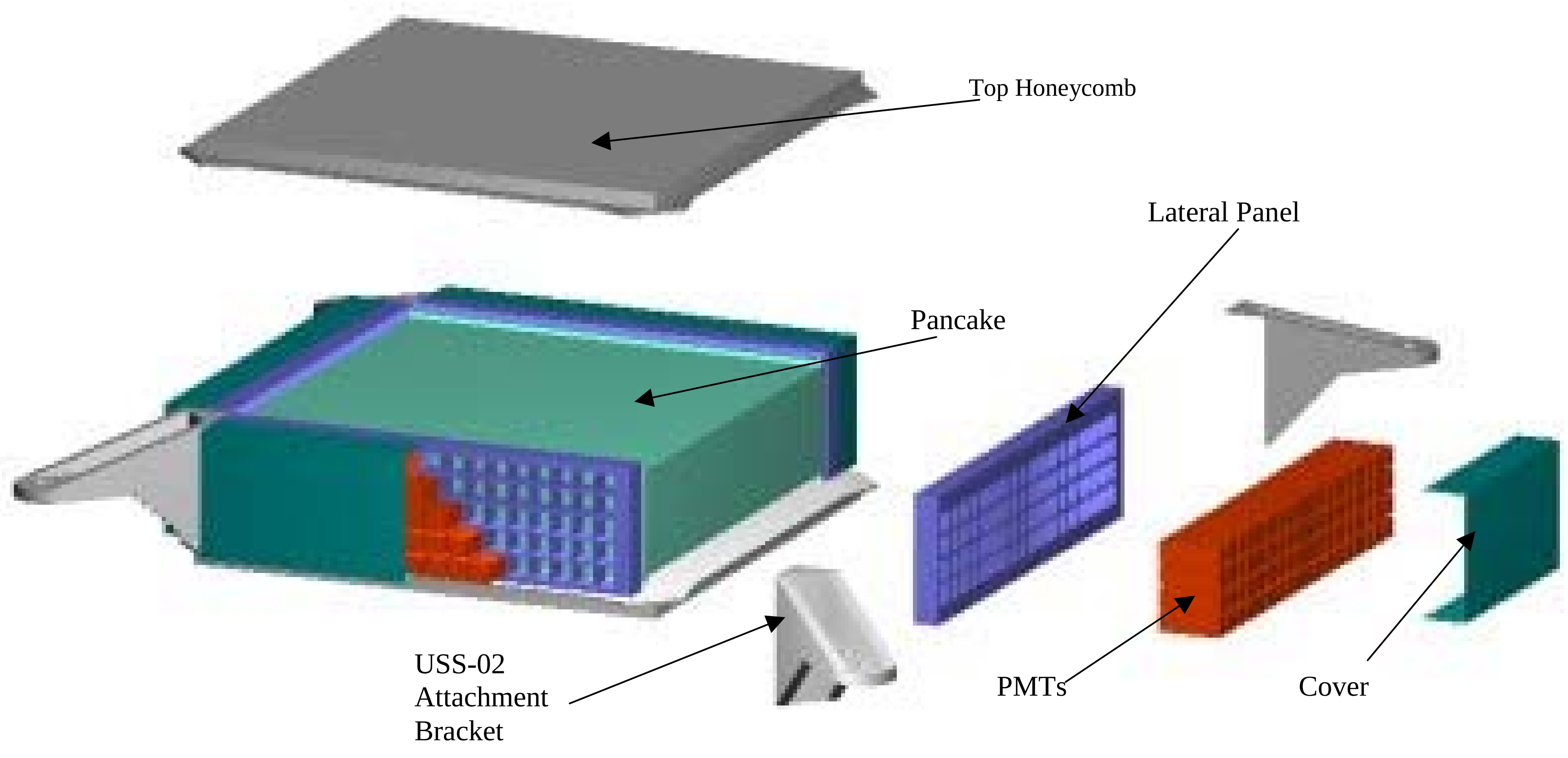


Figure 5.7.2 – Main Parts in ECAL Structure

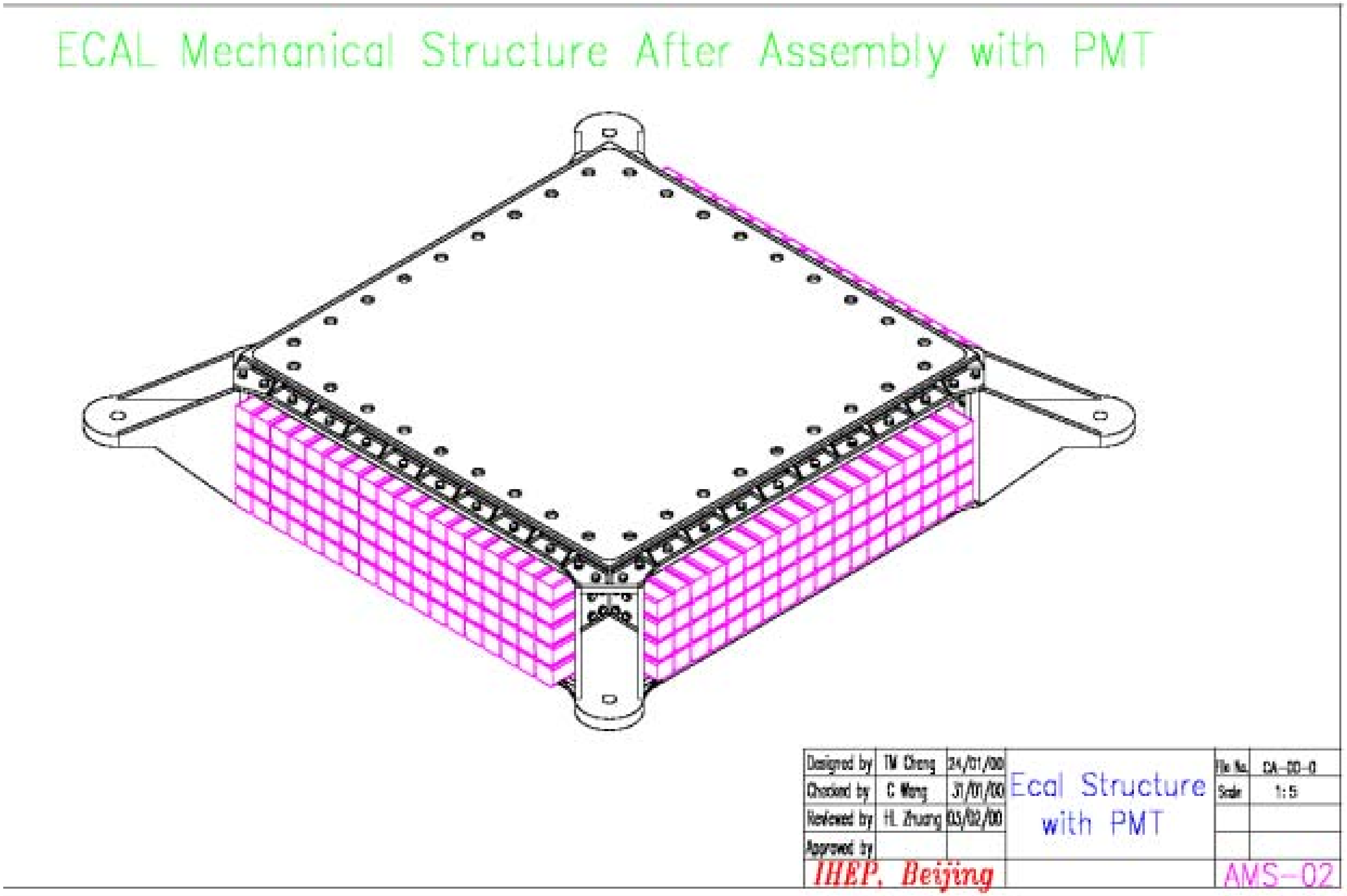


Figure 5.7.3 – ECAL Mechanical Structure After Assembly with PMTs

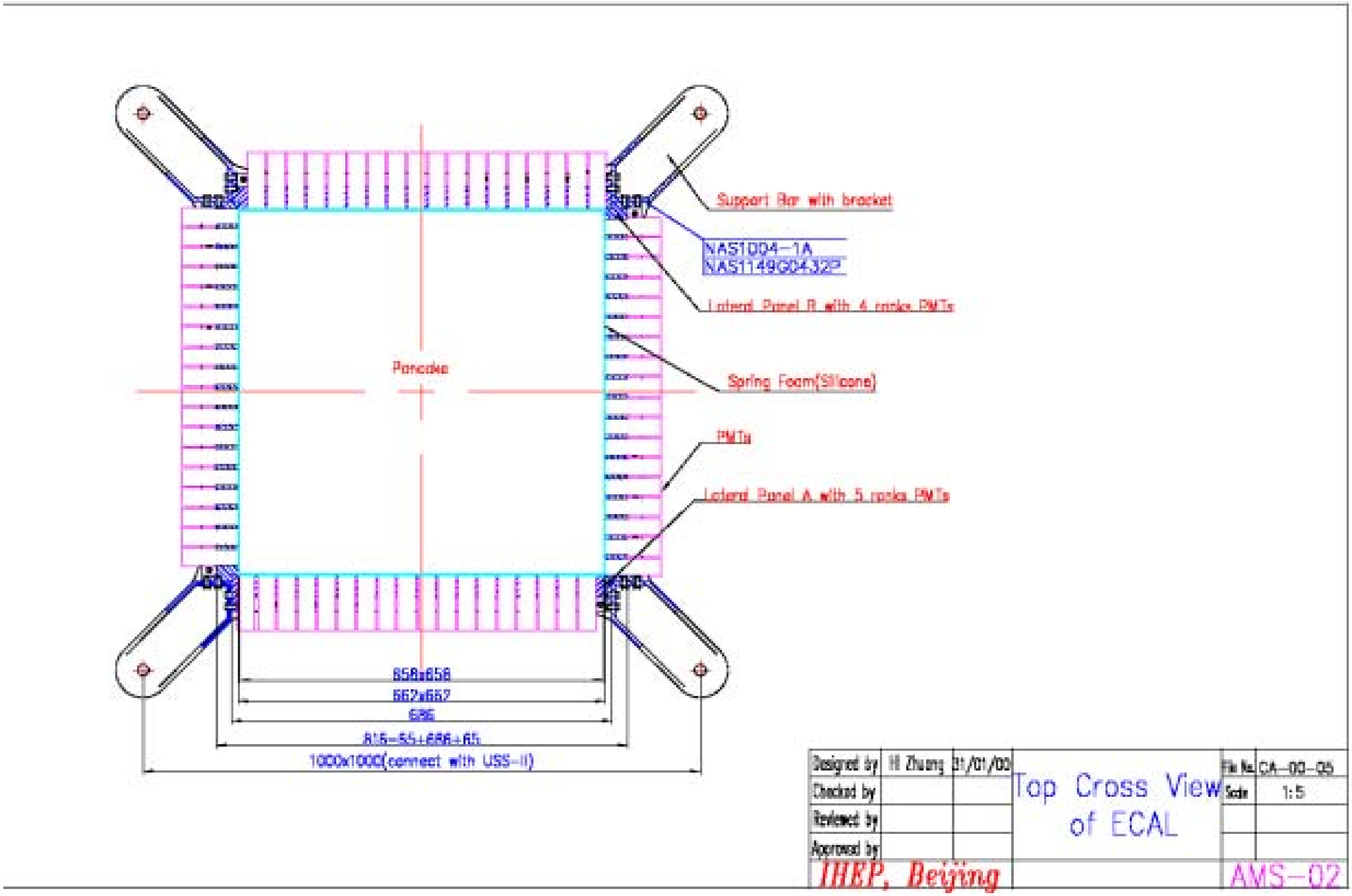


Figure 5.7.4 – Top Cross View of ECAL

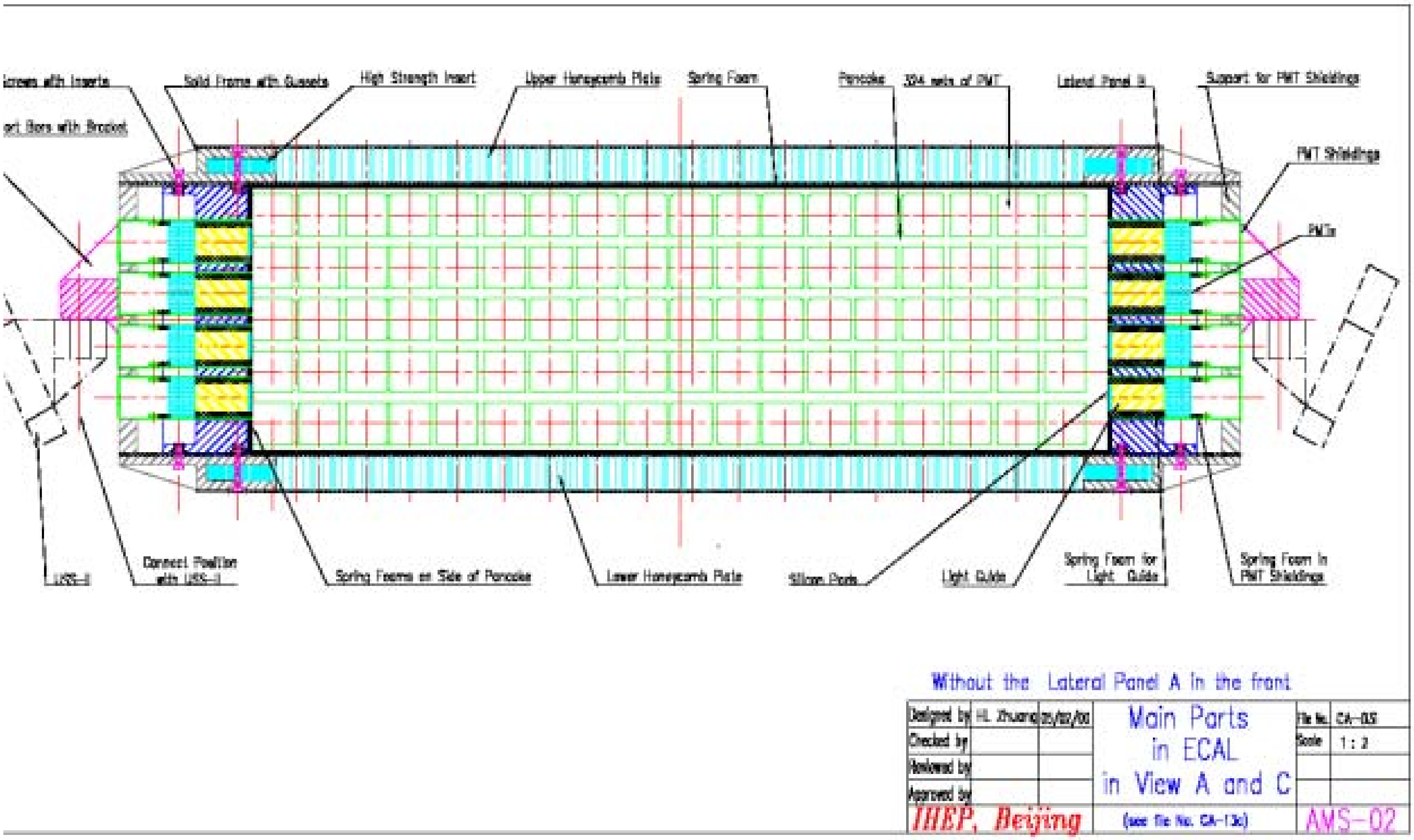


Figure 5.7.5 – Main Parts in ECAL in View A and C

5.8 DATA & INTERFACE ELECTRONICS

The data and interface electronics for the AMS-02 experiment will be housed in electronics crates on the outside of the Unique Support Structure-02 (USS-02). The electronics crates will have aluminum covers. The data and interface electronics will enable the connection of the AMS-02 experiment to the ISS data system.

5.9 ELECTRICAL CABLES

The AMS-02 will be equipped with an ISS provided passive Umbilical Mechanism Assembly (UMA) which will be mated to the ISS active UMA when the AMS is installed on the ISS. Mission Integration provided cables will run from the umbilical connector on the ISS to an input connector on the AMS-02 instrument for both data and power cables. This will be to provide the interfacing between the AMS-02 data/interface electronics and the ISS data systems, and the AMS-02 experiment and the ISS electrical power system via the ISS umbilical connector.

Some AMS-02 Mission Integration provided electrical cables may be required in the pressurized module to provide data interfacing between the ACOP and the AMS-02 instrument located on S3. The ACOP will house the hard drive recorder and other data interfaces for the AMS-02.

5.10 MONITORING AND CONTROL COMPUTERS (MCCs)

Two Monitoring and Control Computers (MCCs) are mounted on the USS-02 or the electronics racks. The MCCs provide the primary data interface between the AMS-02 Experiment low rate data system and the ISS 1553 data bus. The ISS 1553 data bus provides the housekeeping data from the AMS-02 Experiment and command capability to the AMS-02 Experiment through the ISS for transmission to the ground

5.11 POWER DISTRIBUTION BOX (PDB)

The AMS-02 Power Distribution Box (PDB) is mounted on the USS-02 near the passive UMA. The purpose of the PDB is to provide the power interface circuitry between the ISS and the AMS-02. The PDB receives 124v dc power from either or both of the ISS power buses and converts this voltage to 28v dc for distribution to the various AMS-02 subsystems, assuring compliance to the power requirements of SSP 57003 (See Figure 5.11.1). The PDB also distributes the 124v dc power to the AMS-02 Magnet Cryo Avionics Box (CAB). In addition, the PDB provides +5v and +/- 15v dc unswitched power for the AMS-02 mission critical systems, controllers, computers, monitoring circuits, etc. The AMS-02 avionics interfaces block diagrams and interconnect diagrams with the STS and ISS are shown in Figures 5.11.2 thru 5.11.5.

Figure was too large for this document, please find as separate submittal.

Figure 5.11.1 – AMS-02 Power Schematic (updated)

FIGURE 17: AMS EXPERIMENT / SHUTTLE INTERCONNECT DIAGRAM

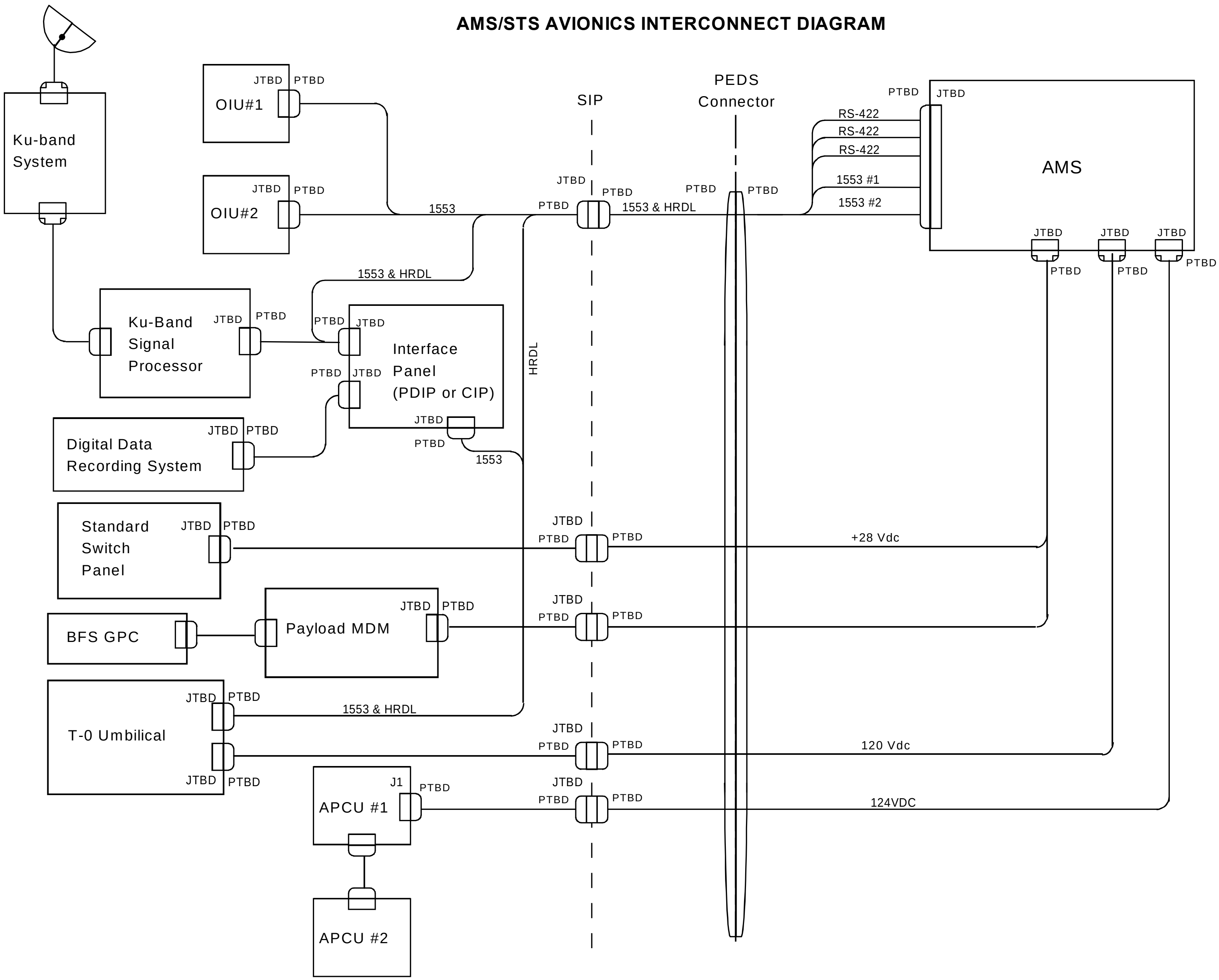


Figure 5.11.2 – AMS/STS Avionics Interfaces Block Diagram

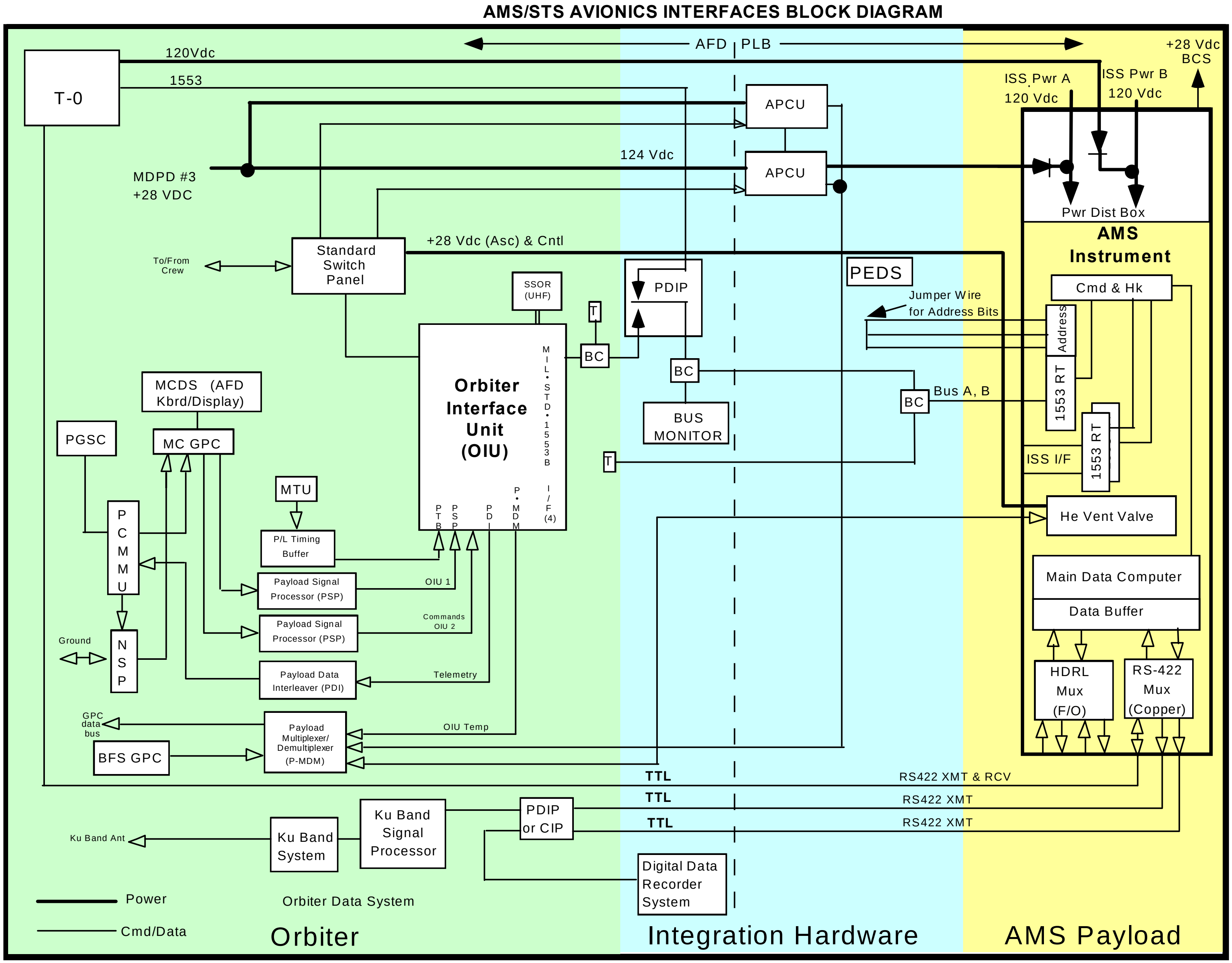


Figure 5.11.3 – AMS/STS Avionics Interconnect Diagram

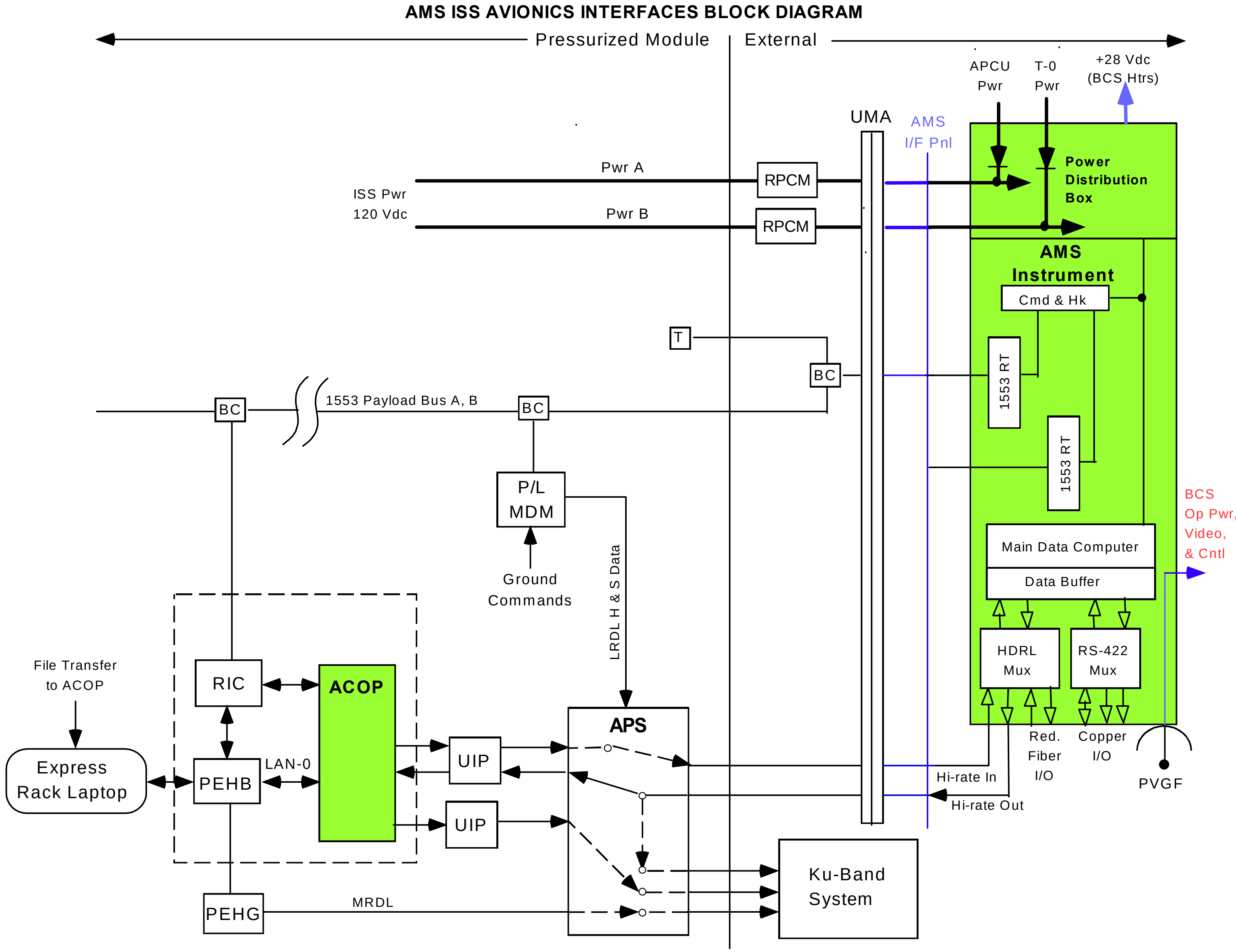


Figure 5.11.4 – AMS/ISS Avionics Interfaces Block Diagram

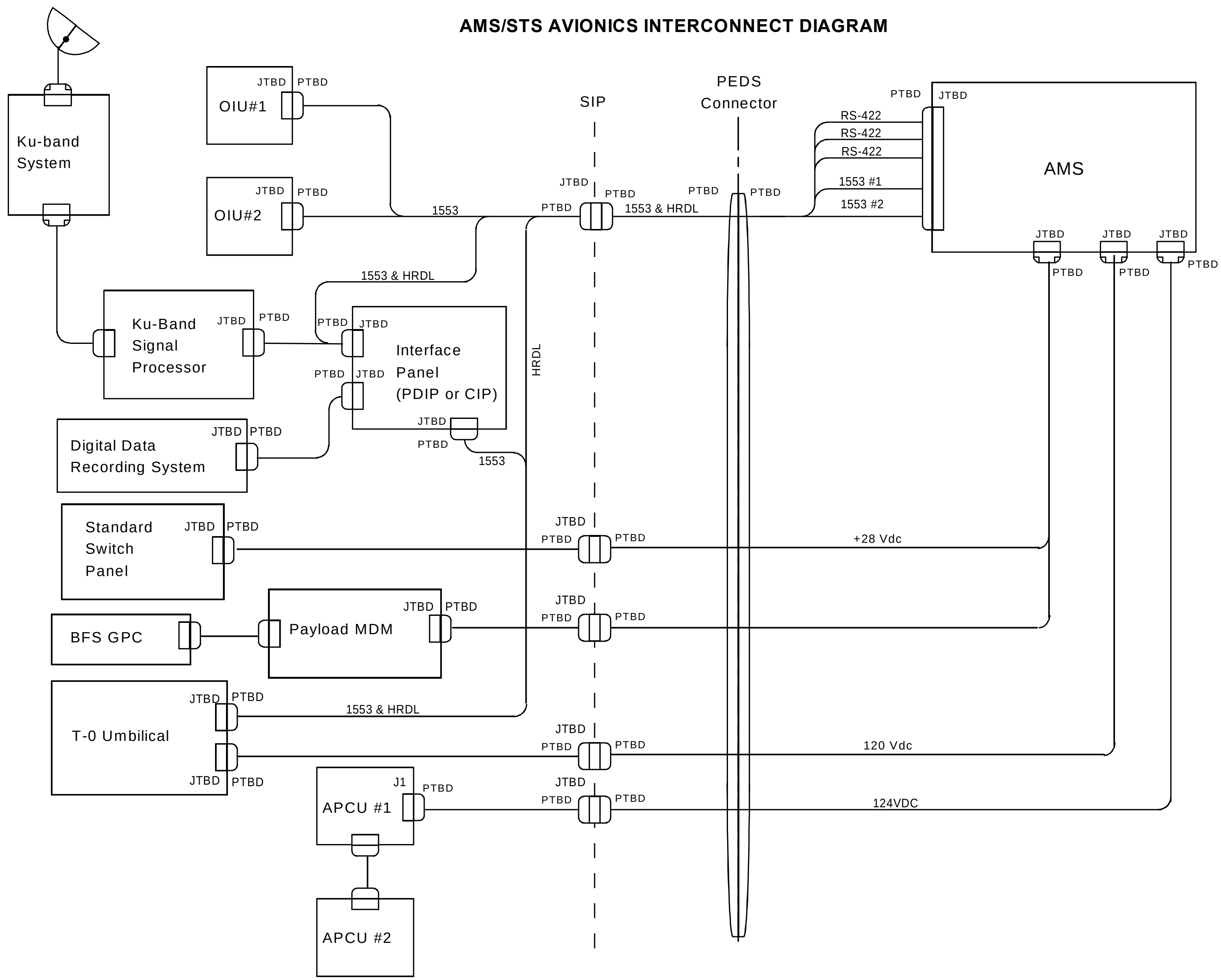


Figure 5.11.5 – AMS/ISS Avionics Interconnect Diagram

5.12 CRYOMAGNET AVIONICS BOX (CAB)

The Cryomagnet Avionics Box (CAB) is mounted between the horizontal and diagonal members of the USS-02 near the sill trunnion. The purpose of the CAB is to provide power to the Charging Circuitry in the Magnet as well as to provide control and monitoring circuitry for charging/discharging operations. The CAB receives unswitched 124v dc power directly from ISS Bus A via the PDB. Magnet charging operations depend upon the operation of Bus A from the ISS. Isolation from the magnet is performed by a transformer contained within the CryoMagnet Current Source (CCS) contained within the CAB, assuring compliance with SSP 57003. All other power requirements in the CAB are supplied from the already isolated PDB supplied 28v dc sources.

The CAB also controls the discharge circuitry for the magnet. Controlled discharges are commanded either by automatic control circuitry or crew/ground commanding. The commanding causes opening of the Magnet persistent switch which then routes the current to a set of 18 rectifiers (shown as the flywheel diode assembly in the AMS power diagram). These rectifiers cause the discharge of the magnet and convert the energy into heat. Each rectifier is capable of dissipating 200 Watts. The magnet discharges in approximately 90 minutes and each rectifier will remain below 212 F (100 deg C). Cages are being developed to ensure that the astronauts cannot access these high temperature items.

The control system for the discharge circuitry will require an Un-interruptable Power Source (UPS). This will allow AMS to perform a quench or a controlled run-down of the magnetic field in the event of loss of commanding or loss of power for an extended (unplanned) period. This UPS system will consist of either a battery or a capacitor bank located on the external payload. The details of this UPS system are being developed at this time; however, if a battery is utilized, it will be designed to meet the requirements documented in JSC-20793, Manned Space Vehicle Battery Safety Handbook as well as NSTS 1700.7B and the ISS Addendum.

5.13 AMS CREW OPERATIONS POST (ACOP)

The AMS Crew Operations Post (ACOP) system consists of a 4PU EXPRESS rack payload drawer assembly (See Figures 5.13.1 thru 5.13.3) and one middeck locker (can be a soft stowage bag) of hard drive media, payload provided external cables, and spares. ACOP serves as a management system for the AMS-02 science data as well as a crew operations post. ACOP is capable of simultaneously receiving, processing, and downlinking the AMS-02 science data stream as provided on the High Rate Data Link (HRDL). Crew control is provided at a low level (power state functions) via the front panel interface. PCS based applications software will provide a robust operations and monitoring ability via a network session with ACOP (See Figure 5.13.4). The ACOP power schematic is shown in Figure 5.13.5.

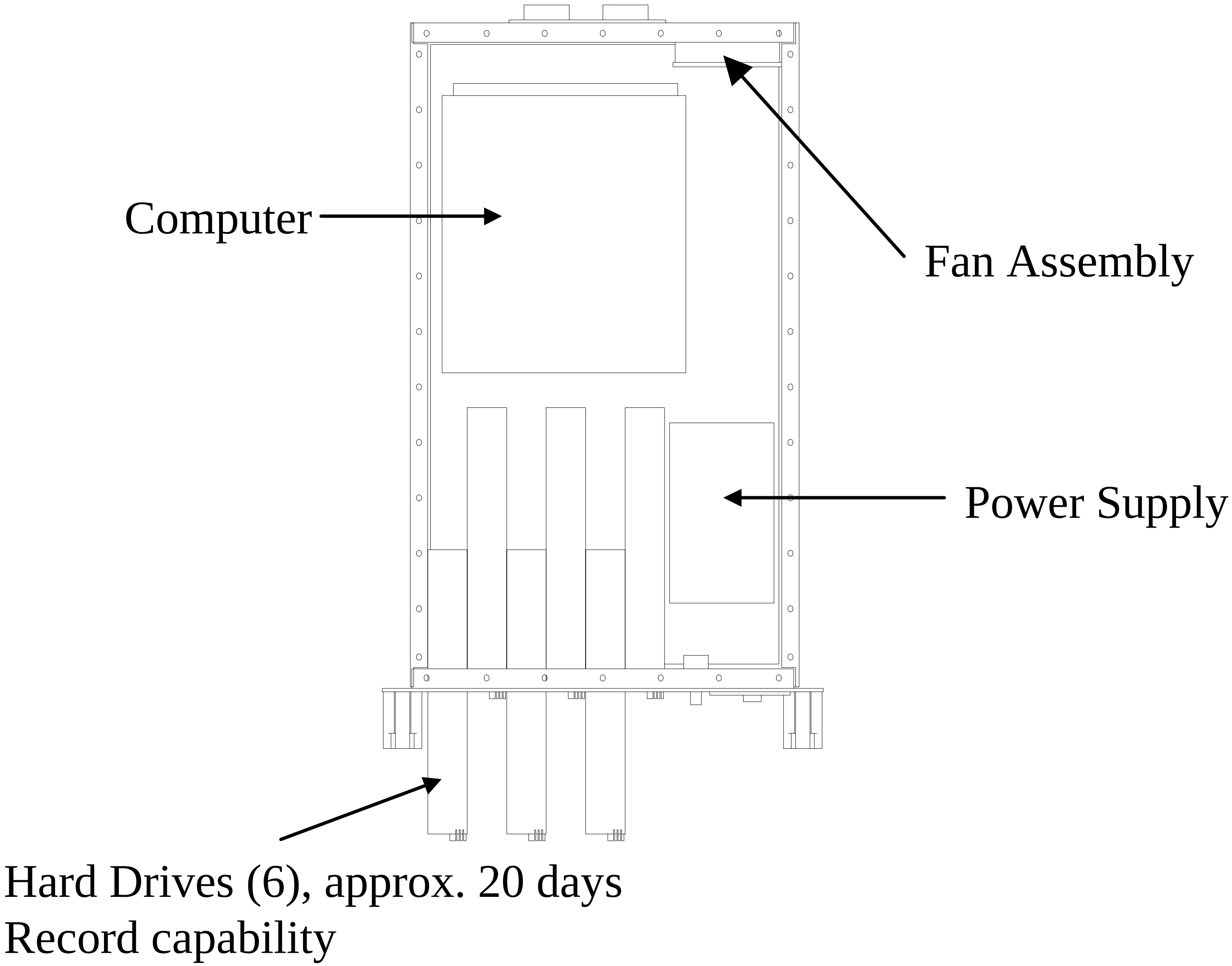


Figure 5.13.1 – ACOP Proposed Drawer Layout (Sheet 1 of 3)

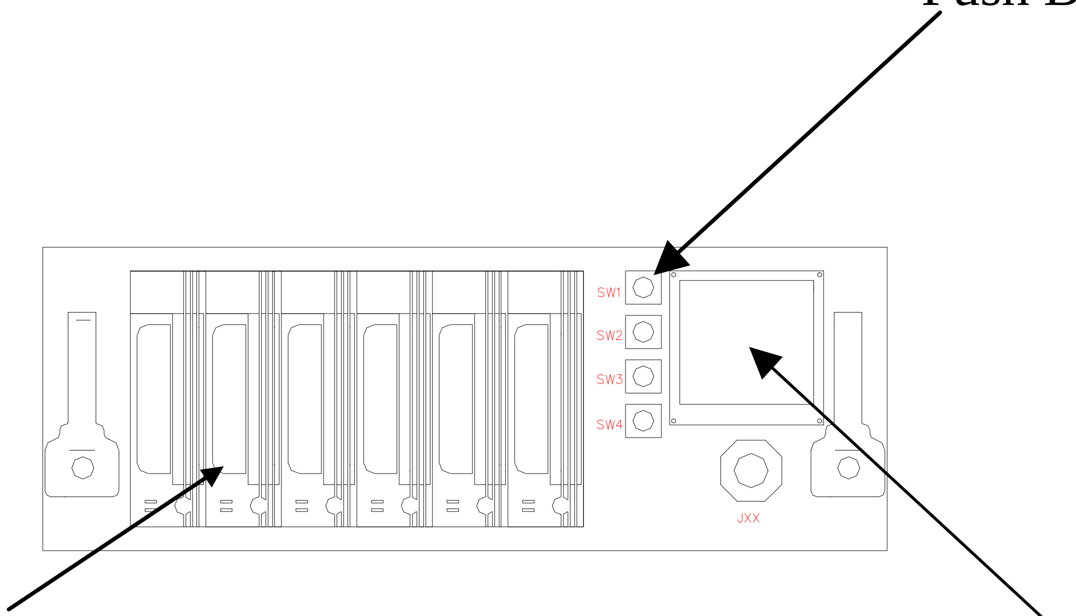


Figure 5.13.2 – ACOP Proposed Drawer Layout (Sheet 2 of 3)

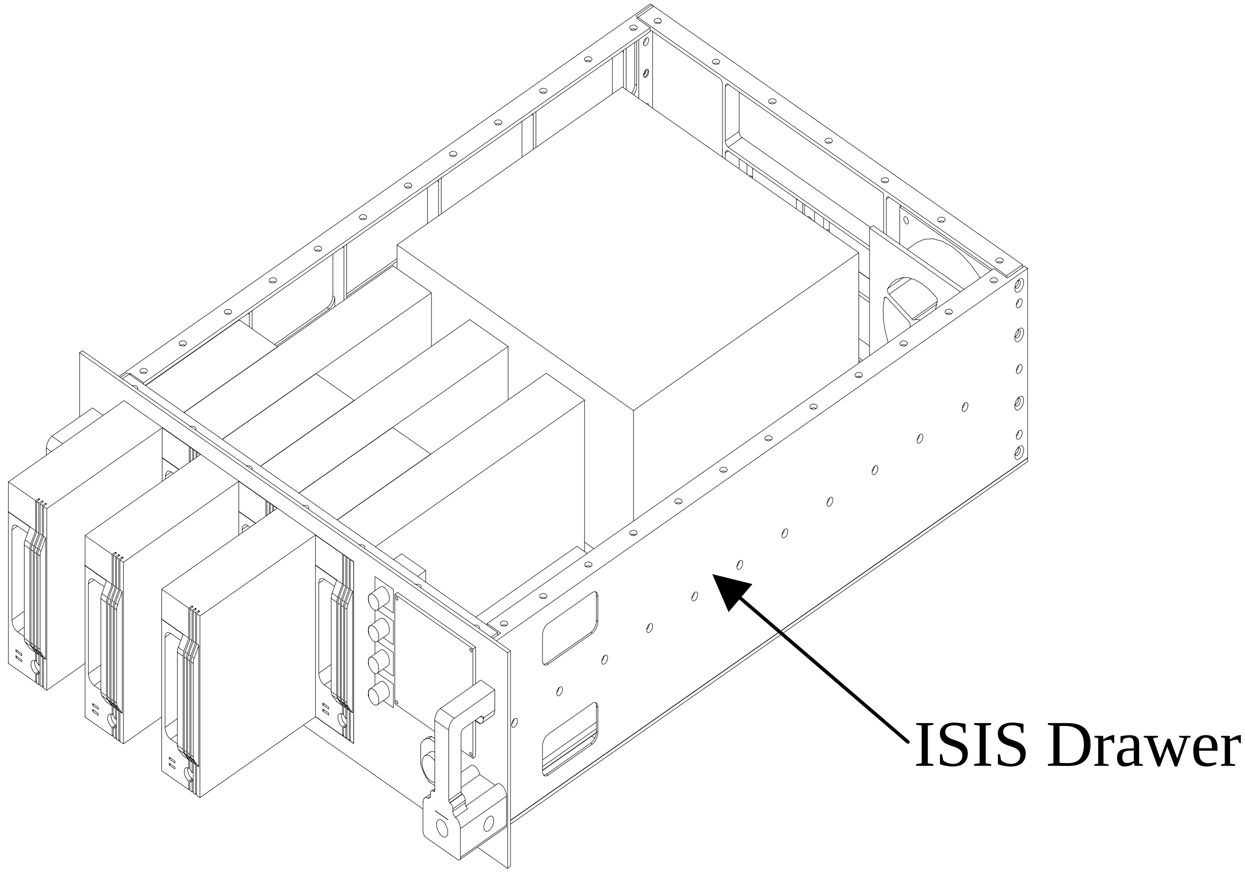


Figure 5.13.3 – ACOP Proposed Drawer Layout (Sheet 3 of 3)

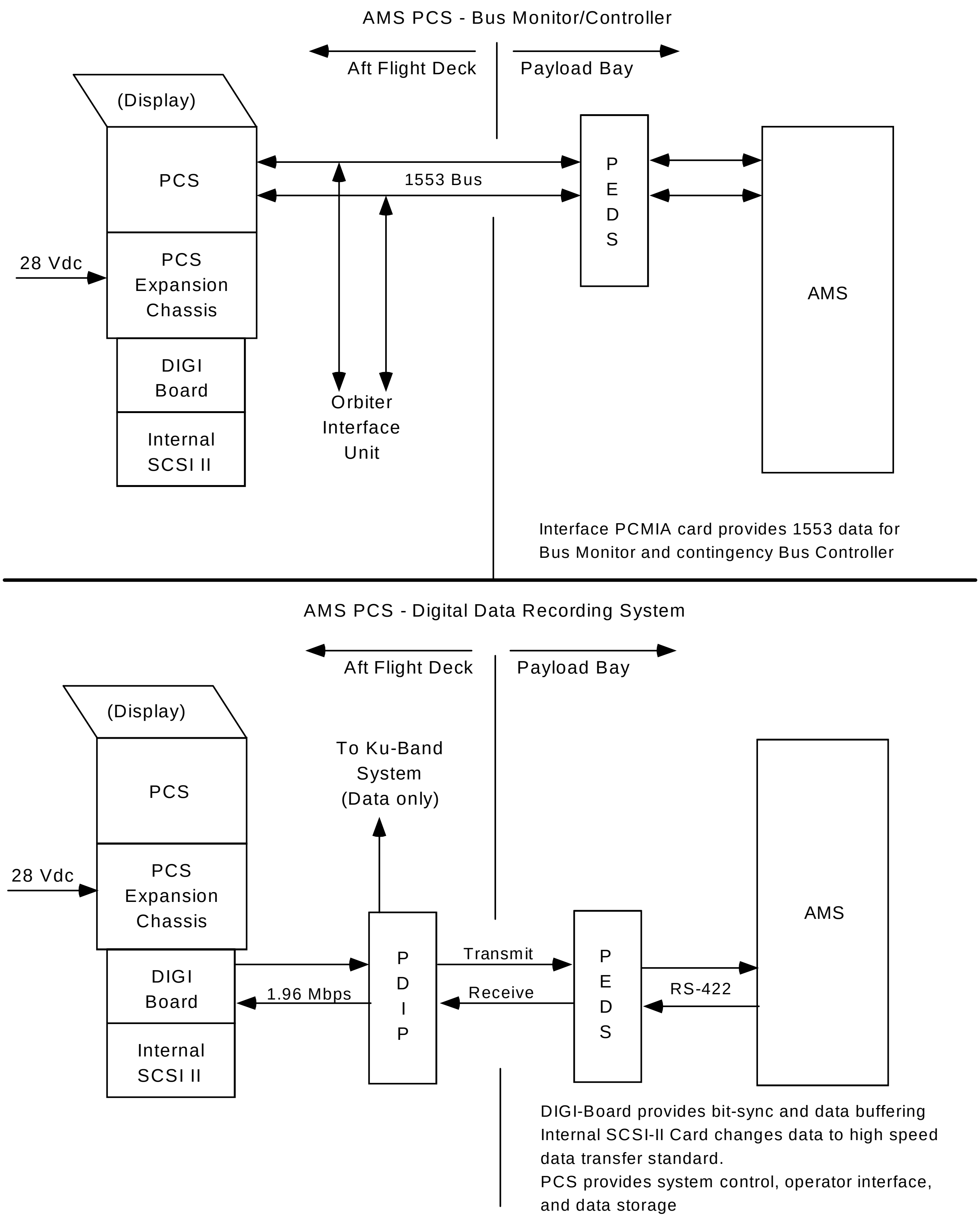


Figure 5.13.4 – AMS PCS – Bus Monitor/Controller

Figure was too large for this document, please find as separate submittal.

Figure 5.13.5 – ACOP Power Schematic

Mounted within the payload drawer are six standard size 5¼ inch hot swap hard drive bays, a power supply, and operations interface consisting of a graphic LCD display and push buttons, a drawer cooling fan, and a Compact PCI card cage assembly. The Compact PCI card cage will be populated with a TBD single board computer, a TBD high data rate link adapter, and a TBD operations interface. The single board computer will provide 100baseT Ethernet, and/or 10baseT Ethernet, serial ports, and SCSI interfaces as appropriate. The single board computer will contain a small battery which will be used to maintain time synchronization and bios parameters. Specifications for this battery are TBD since the specific board has not yet been selected.

The Compact PCI card cage, single board computer, power supply, operations interface, and drawer cooling fan are expected to be COTS equipment, suitable for space use.

The drawer cooling fan will be a 4.5 inch diameter shrouded fan. This fan is a ball bearing type fan and rotates at a maximum speed of 5200 rpm.

The ACOP drawer is attached to the ISS High Rate Data Link (HRDL) by a payload provided fiber Y-cable. This cable attaches via a front panel connector on the ACOP and is routed to the Utility Interface Panel (UIP) for connection to two J7 connectors.

The stowage volume will contain twenty hard drives, payload provided data cables, and spares.

The ACOP will be used as a display/controller for the AMS-02 payload and the science data will be recorded on the hard drives. The science data will be down linked via the KU band system as available. In the event of data congestion on the ISS KU band system, the removable drives can be returned via the STS. The hard drive recorder system has the capability to accept six removable hard drives, of which only one will be active and powered at a time. The six drives, expected to be more than 100 Gbytes each, will provide storage for approximately 20 days of 1.96 Mega-bits per second high rate AMS-02 data without crew intervention. Should the HRDL system become congested, the crew will exchange the hard drives for ones provided in the storage bag.

The hard drives and hard drive hot swap bays are commercial hardware. Comparable hardware was flown on STS-91 for the AMS-01 precursor mission. The plastic covers for the hard drive hot swap bays are being replaced with metal ones (except for the front faces). Each hard drive contains a 3.75" diameter disk that weighs 23 grams and rotates at a maximum speed of 7200 rpm.

Each hard drive hot swap bay contains one 1.4 inch diameter shrouded fan for cooling. This fan is a ball bearing type fan and rotates at a maximum speed of 6500 rpm.

5.14 THERMAL CONTROL SYSTEM (TCS)

The AMS-02 Thermal Control System (TCS) design is currently being developed by the AMS experiment team. Preliminary indications are that this system will consist of radiators mounted either directly to the AMS-02 electronics or mounted on the USS-02. Total radiator surface area could be between 107 and 161 ft^2 (10 and 15 m^2). One or more cooling loops will be used to transport heat from various electronic boxes to the radiators. It is probable that these loops will be driven by a pump and controlled with valves. Working fluids being considered are carbon dioxide (CO_2) and ammonia (NH_3). Other components being considered are capillary pumped loops and heat pipes. These also would probably use ammonia as a working fluid. Heaters will also undoubtedly be required on various experiment components, but these have yet to be defined. Standard NASA Multilayer Insulation (MLI) thermal blankets will also be used.

5.15 Meteoroid and Orbital Debris Shielding

The MOD will be designed, analyzed, built and integrated by NASA/LM Mission Management Office. The shielding is designed to protect the pressure systems on the AMS-02 experiment. These systems include the cryomagnet system including the warm Helium tank, the TRD Gas System, and possibly the TCS. The shielding will be made from various components in different locations depending on the required shield thickness, shape and size. Much of the shielding will be thin aluminum plates with small standoffs from other AMS-02 experiment hardware.

The NASA Hypervelocity Impact Technology Facility has been and will continue to perform all of the analysis and testing for the MOD requirements. Testing will be performed to ensure that the correct ballistic limit equations are used in the analysis. The shields will be designed to meet the ISS and STS requirements.

5.16 Passive Payload Attach System

The passive PAS (see Figure 5.16.1 thru 5.16.5) will be designed, analyzed, built and integrated by NASA/LM Mission Management Office. The passive PAS is the AMS-02 interface to the active PAS on the S3 Truss Segment of ISS. It is designed to sit in the active PAS and react the loads from the active PAS Capture Latch Assembly.

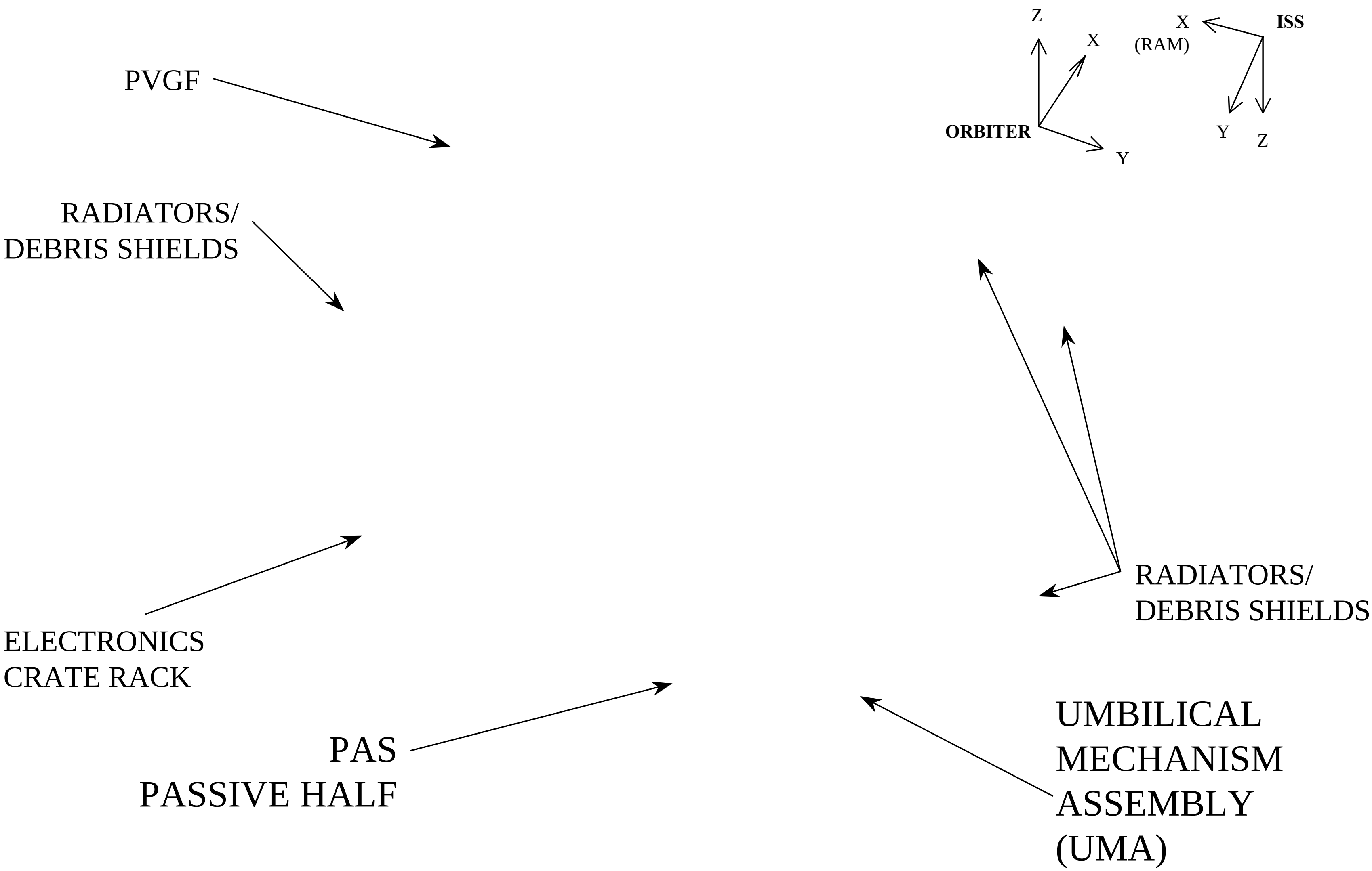


Figure 5.16.1 – USS-02 with Passive PAS & Passive UMA

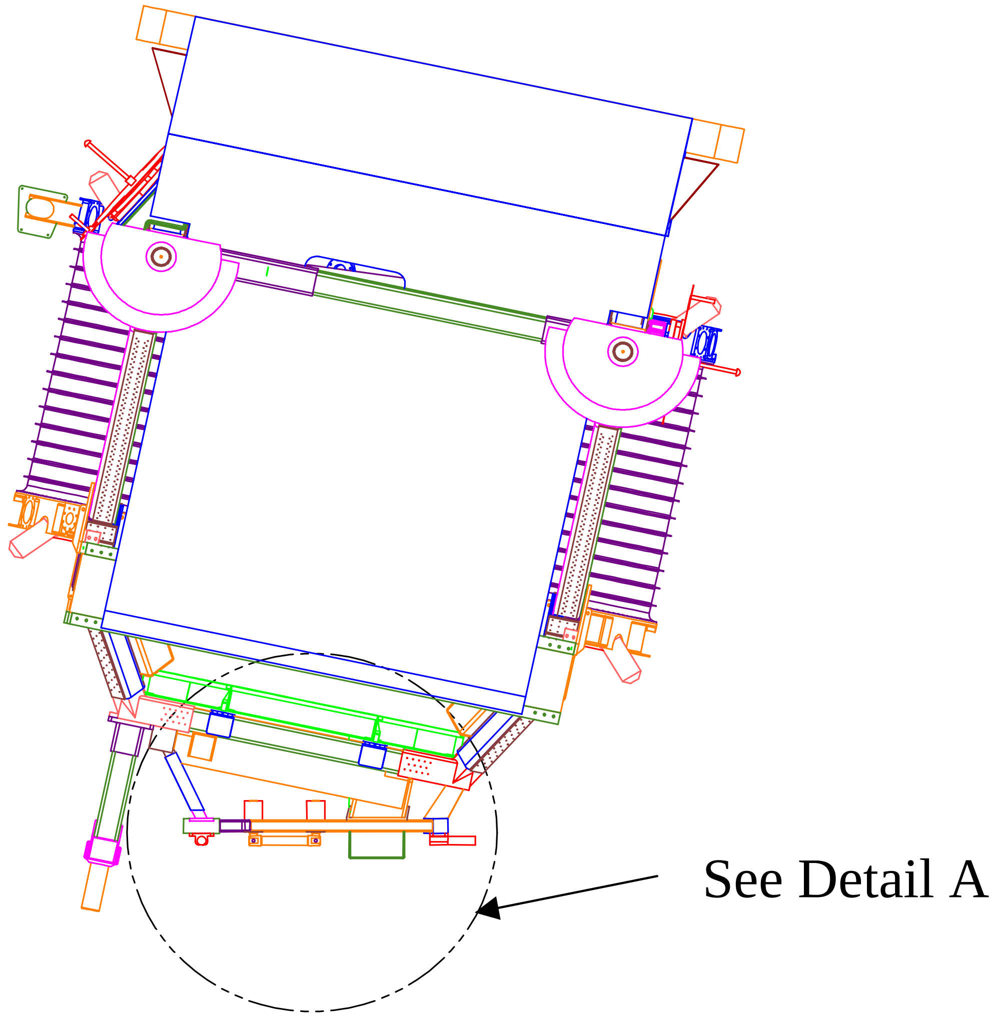
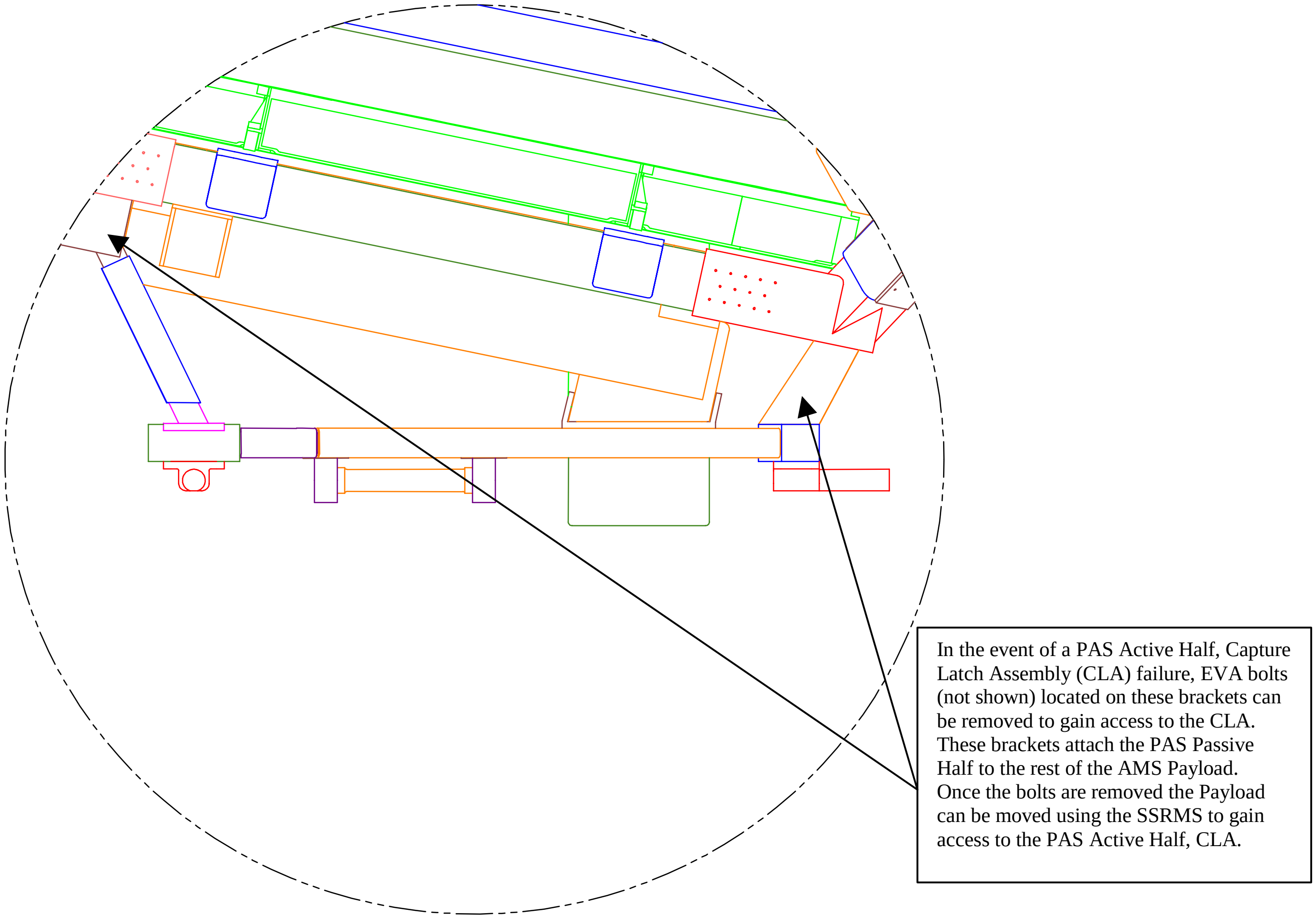


Figure 5.16.2 – Passive PAS (Sheet 1 of 4)



Detail A – No-Spring Design (with rigidly attached capture bar)
-Design of capture bar is TBD Based on ISS Requirement Changes

Figure 5.16.3 – Passive PAS (Sheet 2 of 4)

Springs

Detail B – Spring Design (with spring loaded capture bar)
-Design of capture bar is TBD Based on ISS Requirement Changes

Figure 5.16.4 – Passive PAS (Sheet 3 of 4)

PAS Attached to ISS
(AMS/USS removed for clarity)

Figure 5.16.5 – Passive PAS (Sheet 4 of 4)

5.17 SPACE SHUTTLE PROGRAM (SSP) AND ISS PROGRAM PROVIDED HARDWARE

The Space Shuttle Program (SSP) provided hardware that will be used with the AMS-02 includes: Two Orbiter Interface Units (OIU's), two PGSCs with expansion assemblies and power cables, Flight Releasable Grapple Fixture (FRGF) and Remotely Operated Electrical Umbilical (ROEU) or Passive Electrical Disconnect System (PEDS). The AMS-02 payload will also require the use of the Shuttle Remote Manipulator System (SRMS).

The two OIU's (one will be a backup) will be used as 1553 bus controllers and will serve as the uplink/ downlink interface for housekeeping data and commands. The OIU's will be mounted in the Payload Station L11 Console in the Orbiter aft flight deck, as designed. One SSP provided Personal Computer System (PCS) and SSP provided 1553 data bus card will be used as a display for the crew for AMS housekeeping data and for contingency loading of software.

A FRGF on the AMS-02 payload will be used by the SRMS to lift the AMS-02 out of the Orbiter payload bay. The mounting details for the FRGF are currently TBD.

The ROEU or PEDS will be used to make the electrical interface between the Shuttle and the AMS-02 payload. The PEDS or ROEU will be mounted to the USS-02 near the primary starboard trunnion.

The ISS Program provided hardware that will used with the AMS-02 includes: Two Assembly Power Converter Units (APCU’s), Power Video Grapple Fixture (PVGF), active PAS, passive Umbilical Mechanism Assembly (UMA), active UMA, Berthing Cues System (BCS), 4 Panel-Unit (PU) International Subrack Interface Standard (ISIS) drawer in the ISS EXPRESS Rack. The AMS-02 payload will also require the use of the Space Station Remote Manipulator System (SSRMS).

The APCU's (one will be a backup) will be used to supply 124v dc power to the AMS. Two APCU's are being used to provide redundancy. They will be mounted in bay 5 on the port side of the Orbiter, as designed, per the APCU Interface Control Document (ICD).

A PVGF on the AMS-02 payload will be used by the SSRMS to place the AMS-02 on to the truss attach site. The mounting details for the PVGF are currently TBD by ISS.

The passive UMA will be used to electrically attach the AMS-02 payload to the ISS truss attach site. The UMA attaches to the lower USS-02.

The BCS is a camera and avionics package that is electrically connected to the PVGF. The system will be used during the berthing operation to the ISS truss attach site. The BCS attaches to the lower USS-02.

The 4 Panel-Unit (PU) International Subrack Interface Standard (ISIS) drawer in the ISS EXPRESS Rack will be used for the ACOP.

The active PAS and UMA are provided by ISS as part of the S3 truss assembly.

The AMS-02 payload is using all of the SSP and ISS provided hardware for the same purposes that they were designed.

6.0 AMS FLIGHT OPERATIONS SCENARIO

The AMS-02 will require power during launch. The SFHe tank nominal vent valve must be opened once the atmospheric pressure falls below 20 mbar. This vent rate into the payload bay has been approved by Shuttle Integration. This is required for mission success to ensure mission required Helium endurance. 28 Vdc momentary power will be applied to a solenoid that operates the valve to perform this function. In the event of an abort, this same vent valve must be closed when descending through ~100,000 ft (atmospheric pressure <20 mbar). The trigger to open this valve is a barometric switch, and as a backup, a time-tagged command via Backup Flight System General Purpose Computer will be used. On-orbit, by an Mission Elapsed Time (MET) of 2 hrs. 30 mins., the APCUs and cryocoolers will be activated and housekeeping data will be available. Then the AMS-02 avionics subsystems will be activated and checkout operations will be performed. No magnet charging will be performed while in the Orbiter. After docking with the ISS, the AMS-02 will be powered down prior to transfer operations. Transfer to the ISS will be by MET day 4. The transfer operations will be an SSP and ISS Program provided service using SSP and ISS provided hardware. The SRMS will first grapple the FRGF. Then, after the AMS-02 has been disconnected from the ROEU or PEDS, and the Orbiter payload retention latches are opened, the AMS-02 will be lifted out of the Orbiter payload bay. The ACOP will be powered up at this time. The SRMS will handover the AMS-02 to the SSRMS. The SRMS will transfer the AMS-02 to the S3 attach site without moving the Mobile Transporter. Mechanical and electrical attachment to the S3 site will be made via the PAS and UMA. The AMS-02 avionics will then be powered up. Following checkout, the magnet charging operations will begin (with crew monitoring). Once charging operations have been completed, science data acquisition will begin. Primary control of the AMS-02 will be from the ground. Crew interfaces to the AMS-02 include the ACOP Liquid Crystal Display (LCD), switches and the ISS EXPRESS rack laptop.

7.0 SAFETY DISCUSSION

A safety analysis was performed for the AMS-02 to identify potential hazards. Hazards identified which have standard controls are documented on the Flight Payload Standardized Hazard Control Report, JSC Form 1230, in Appendix A. Unique hazards identified are documented on the Payload Hazard Reports, JSC Form 542B, in Appendix B.

7.1 STRUCTURAL FAILURE OF SEALED AND VENTED CONTAINERS

The 44 TRD segments, the TRD small mixing tank and the four cryocoolers' pump housings have been identified as sealed containers.

Each one of the 44 TRD segments is protected from overpressurization by two automatic control valves located upstream of these segments in series. Structural and fracture control assessments will be performed to ensure that the requirements of NASA-STD-5003 are met. The materials used to manufacture these sealed containers will be reviewed and approved by JSC EM2/Materials and Process Technology Branch. The proof pressure tests and the leak tests will be performed for these containers as established in the AMS-02 Structural Verification Plan (SVP), JSC – 28792A.

The whole RICH assembly and the TRD octagon enclosure have been identified as vented containers. Also, making the TOF and Si Tracker assemblies light tight will create a vented container. A structural assessment will be performed to ensure that the vents are properly sized to minimize the pressure differential during ascent and descent, and to ensure that the factors of safety – 1.4 for the shuttle and 1.5 for the ISS – are not compromised due to the pressure loads resulted from the vents. The vent sizes will be verified by the review of drawings and certification of the as-built hardware to the approved drawings.

The sealed and the vented containers are addressed on the JSC Form 1230 in Appendix A.

7.2 SHARP EDGES

The AMS-02 Orbiter crew compartment hardware and ISS module hardware will be designed and inspected to meet the sharp edge requirements of NSTS 07700, Vol. XIV, Appendix 9, "System Description and Design Data - Intravehicular Activities", and SSP 57000, "Pressurized Payload Hardware Interface Requirements Document". The AMS-02 Orbiter payload bay hardware and externally mounted hardware on the ISS will be designed and inspected to meet the sharp edge requirements of NSTS 07700, Vol. XIV, Appendix 7, "System Description and Design Data - Extravehicular Activities". This is addressed on JSC Form 1230 in Appendix A.

7.3 SHATTERABLE MATERIALS

The AMS-02 Experiment TOF counters, RICH and ECAL have Photo Multiplier Tubes (PMTs) with glass windows. The PMTs are in housings which will contain the glass if broken. This will be verified by review of drawings and certification of flight hardware conformance to drawings. This is addressed on JSC Form 1230 in Appendix A.

7.4 FLAMMABLE MATERIALS

For the JSC built AMS-02 hardware, the materials will be selected per the Materials and Processes Technology Information System (MAPTIS). For the non-JSC built AMS-02 hardware, a flammability assessment will be performed per NSTS 22648, "Flammability Configuration Analysis for Spacecraft Applications". Verification will be by review of materials lists, review of drawings and inspection of hardware. This is addressed on JSC Form 1230 in Appendix A.

7.5 ELECTROMAGNETIC INTERFERENCE (EMI)

With the exception of the magnetic field, the AMS-02 design will comply with the EMI requirements of NSTS 07700, Vol. XIV, Attachment 1, NSTS-21000-IDD-ISS, Rev. Current, "ISS Interface Definition Document", and SSP 30237B, "Space Station Electromagnetic Emission and Susceptibility Requirements". The AMS-02 will be tested at the subsystem level per MIL-STD-462, "Military Standard, Measurement of Electromagnetic Interference Characteristics". To show compliance with the Space Shuttle and ISS requirements, the AMS payload will provide the required electromagnetic data specified by NSTS 21288, "Required Data/ Guidelines for Payload/Shuttle Electromagnetic Compatibility Analysis", for review by MS2/Integration Engineering Office. This is addressed on JSC Form 1230 in Appendix A. For the magnetic field, AMS-02 will request a waiver/exceedance from ISS (the magnet is not operated on the STS).

7.6 TOUCH TEMPERATURE

The AMS-02 Orbiter crew compartment hardware and ISS module hardware will be designed to meet the requirements of NASA Letter #MA2-95-048. The AMS-02 Orbiter payload bay hardware and externally mounted hardware on the ISS will be designed to meet the requirements of NSTS 07700, Vol. XIV, Appendix 7. Verification will be by thermal analysis and/or testing. This is addressed on JSC Form 1230 in Appendix A.

7.7 ELECTRICAL POWER DISTRIBUTION CIRCUITRY

To protect the Orbiter and ISS from any AMS-02 payload electrical faults, the AMS-02 electrical power distribution circuitry is being designed to meet the requirements of NASA Letter #TA-92-038, "Protection of Payload Electrical Power Circuits", and SSP 57000. This is addressed on JSC Form 1230 in Appendix A.

7.8 FLAMMABLE ATMOSPHERE

During launch and landing, 28 V power will be available from the Standard Switch Panel (SSP) to operate a solenoid. The purpose of the solenoid is to open a He Vent Valve during launch and close it during landing. An assessment will be performed to ensure that any potential ignition sources during launch and landing are controlled per NASA Letter #NS2/81-M082. The Multilayer Insulation (MLI) blankets will be grounded per NSTS-21000-IDD-ISS to dissipate electro-static discharge (ESD). Proper grounding of the MLI will be verified by the review of drawings, inspection of the as-built hardware and MLI grounding tests. The concern of the ignition of flammable atmosphere in the payload bay is addressed on the JSC Form 1230 in Appendix A.

7.9 ROTATING EQUIPMENT

The AMS-02 rotating equipment consists of: a 4.5 inch diameter, 5200 nominal rpm fan in the ACOP drawer; six Hard Drives (HDs), each with a 3.75 inch diameter, 7200 maximum rpm disk; six 1.4 inch diameter, 6500 maximum rpm fans (one in each HD hot swap bay); two TRD gas system pumps, each with a 5000 maximum rpm, 24 V dc brushless motor; and TBD TCS rotating equipment. All of the AMS-02 rotating equipment is being designed to meet the requirements of NASA-STD-5003, “Fracture Control Requirements for Payloads Using the Space Shuttle”. Verification will be by review of drawings and certification of flight hardware conformance to drawings. This is addressed on JSC Form 1230 in Appendix A.

7.10 MATING/DEMATING POWER CONNECTORS

The only AMS-02 mating/demating of power connectors will be contingency operations for mission success only. The EVA contingency mating/demating will be to change from primary to redundant connectors on an interface panel on the USS-02. This will be required in the event of an ISS bus failure or an AMS-02 mission success failure. The mating/demating operations will meet the requirements of NASA Letter #MA2-99-170, "Crew Mating/Demating of Powered Connectors". This is addressed on JSC Form 1230 in Appendix A.

7.11 CONTINGENCY RETURN AND RAPID SAFING

The AMS-02 contingency return and rapid safing requirements are TBD. The procedures will comply with the requirements of JSC Letter #MA2-96-190, "Contingency Return and Rapid Safing", and will not impede emergency IVA egress to remaining adjacent pressurized ISS volumes. This is addressed on JSC Form 1230 in Appendix A.

7.12 STRUCTURES

The AMS-02 hardware is being designed to positive margins of safety with the factors of safety specified in Appendix A of the AMS-02 Structural Verification Plan (SVP) (JSC-28792A). The verification methods are also specified in JSC 28792A. Structural materials with high resistance to stress corrosion are being selected for the AMS-02 payload per MSFC-SPEC-522B, "Design Criteria for Controlling Stress Corrosion Cracking", wherever possible, or Material Usage Agreements (MUA's) will be submitted for approval. JSC 25863A, "Fracture Control Plan for JSC Flight Hardware", is being used to implement the fracture control requirements of NASA-STD-5003 and SSP-30558B. All #10 (~5mm) fasteners and larger will be tested to ensure compliance with JSC-23642D. Back-off prevention will be used on all safety-critical fasteners. Approved drawings and procedures will be used for manufacturing and assembly. This is addressed on Hazard Report #AMS-02-1 in Appendix B.

7.13 MATERIALS OFFGASSING

Materials which do not offgas toxic products will be selected for AMS-02 from the MAPTIS or MSFC-HDBK-527E/JSC 09604E, wherever possible, or MUAs will be submitted for approval. Verification will be by review of materials lists, review of drawings and inspection of hardware and/or offgassing tests per NASA-STD-6001. Verification will be complete when JSC EM2/Materials and Process Technology Branch issues a Materials Certification Letter. This is addressed on Hazard Report #AMS-02-2 in Appendix B.

7.14 RUPTURE OF VACUUM CASE AND/OR SUPERFLUID HELIUM TANK/LINES/ FITTINGS/PRESSURIZED COMPONENTS

The AMS-02 hardware is being designed to positive margins of safety with the factors of safety and MDP determinations specified in Appendix A of the AMS-02 Structural Verification Plan (SVP) (JSC-28792A). Verification will be by analyses and tests as specified in JSC-28792A.

The SFHe tank and vacuum case each have three burst disks in series and the Cold Buffer Volume has one burst disk. Verifications for the burst disks will be by review of drawings and testing.

Structural materials with high resistance to stress corrosion are being selected for the AMS-02 payload per MSFC-SPEC-522B, "Design Criteria for Controlling Stress Corrosion Cracking", wherever possible, or Material Usage Agreements (MUA's) will be submitted for approval.

Compatibility of the SFHe wetted components with He will be verified by review of the final materials list.

Approved drawings and procedures will be used for manufacturing and assembly

JSC 25863A, "Fracture Control Plan for JSC Flight Hardware", is being used to implement the fracture control requirements of NASA-STD-5003 and SSP-30558B.

Freezing of He in the SFHe pressure system will be prevented by ensuring that the pressure and temperature parameters within the system can not approach the solid phase. Verification will be by thermal analysis.

Overfilling of the SFHe pressure system will be prevented by using approved ground operations procedures.

All He plumbing lines and pressurized components will be properly sized for the MDP of the respective lines/components. All lines and components will meet the requirements of NSTS 1700.7B, the ISS Addendum and SSP 30559B. Verification will be by analyses and tests as specified in the AMS-02 SVP (JSC-28792A).

The SFHe tank are being designed to meet the Meteoroid and Orbital Debris (M&OD) Probability of Non-Penetration (PNP) requirement of 0.997 for 5 years. This will be verified by M&OD Risk Assessments.

These items are addressed on Hazard Report #AMS-02-3 in Appendix B.

7.15 VENTING OF HELIUM GAS

The AMS-02 helium venting systems design includes deflectors to minimize the vent stream from impinging (pressure and thermal effects will be accounted for) on the AMS-02 structure, the Orbiter, the ISS or other payloads. The venting will be zero-thrust. This will be verified by analysis/test.

The AMS-02 will be designed such that the worst-case venting rates will not cause overpressurization of the payload bay. This will be verified by analysis.

Loss of vacuum around the SFHe pressure system due to leakage of o-rings, welds, mechanical fittings, burst disks (also premature rupture) or puncture in the vacuum case is controlled by design as specified in the AMS-02 SVP (JSC-28792A). The verification methods are also specified in JSC 28792A.

The AMS-02 components will be compatible with helium and any liquid air that may condense on cold surfaces. Drip pans will be located at the points of liquid air impact. Verification will be by review of drawings.

These items are addressed on Hazard Report #AMS-02-4 in Appendix B.

A Venting Analysis Report has been prepared and is enclosed as a separate attachment for the Flight Safety Package.

7.16 RUPTURE OF THE FOLLOWING AMS-02 PRESSURE SYSTEMS: TRD GAS SUPPLIES, WARM HELIUM SUPPLY OR AMS-02 THERMAL CONTROL SYSTEM (TCS)

The above pressure systems are designed to positive margins of safety with the factors of safety and MDP as specified in the AMS-02 Structural Verification Plan (SVP), JSC-28792A. Structural assessments and tests will be performed to verify the structural integrity of these pressure systems.

Each of these pressure systems will be supplied with relief valve(s) which will be tested to ensure that they relieve at or below the MDP of the respective system. These relief valves will be properly sized to ensure that the MDP of the respective pressure system is not exceeded.

The materials chosen for these pressure systems will meet the requirements of MSFC-SPEC-522B. The structural materials with high stress corrosion cracking will be used where possible, and the materials of the wetted parts will be selected to ensure inertness with the gases in contact. A fracture control and material assessment will be performed to ensure the use of proper materials.

For proper manufacturing and assembly, the approved drawings and procedures will be used, and will be verified by certification documentation. Approved ground operations procedures will be used to avoid overfilling of the pressure systems.

The pressure systems will be protected from the M&OD impact by providing properly designed M&OD shields around them. The MOD Risk Assessments are being performed to ensure that the pressure systems meet the Probability of Non-Penetration (PNP) requirement of 0.997 for 5 years.

Rupture of the above pressure systems are addressed in Hazard Report #AMS-02-5 in Appendix B.

7.17 VENTING OF GAS(ES) FROM THE TRD, WARM HE SUPPLY AND/OR TCS

Venting of the gases from the above pressure systems during normal on-orbit operations is very minimal and will not cause high pressure impingement. Potential high pressure impingement due to emergency venting during launch, landing and on-orbit will be eliminated or minimized by proper orientation of the vent lines. Also, these vent lines will be designed to produce zero thrust. Venting of gases from the above pressure systems is addressed in Hazard Report #AMS-02-6 in Appendix B.

7.18 AMS-02 MAGNETIC FIELD

The AMS-02 cryomag has been designed to reduce the magnetic field outside of the magnet as much as possible. Magnetic field susceptibility tests have been performed on the Extravehicular Mobility Unit (EMU), the Simplified Aid for EVA Rescue (SAFER) and the Pistol Grip Tool (PGT) to establish new requirements. The test report is pending. ISS equipment requirements are TBD. Compliance with the requirements will be verified by magnetic field measurements and mapping. This is addressed on Hazard Report #AMS-02-7 in Appendix B.

7.19 ELECTRICAL SHOCK

All AMS-02 high voltages in the AMS-02 payload will be enclosed or covered with MLI and/or light tight material to prevent exposure to EVA crewmembers. All AMS-02 electrical components will be grounded to the USS-02 and to the Orbiter per NSTS-21000-IDD-ISS, and the AMS-02 will be grounded to the ISS through the PAS per SSP 57003. Verification will be by review of drawings, inspection of hardware and grounding tests. This is addressed on Hazard Report #AMS-02-8 in Appendix B.

7.20 EXCESSIVE IONIZING RADIATION

The TRD contains ten calibration tubes with a 0.2µCi deposit on the inside of each tube. The 1 mm thick wall of each tube attenuates the 5.9 keV radiation to a level that is less than detectable. Each tube is mounted inside a stainless steel container. Each container is located in a box on the TRD detector or in Box C. Verification will be by review of drawings, certification of flight hardware conformance to drawings and measurement checks of the radiation levels of the calibration tubes. This is addressed on Hazard Report #AMS-02-9 in Appendix B. JSC Form 44 for the radiation sources is attached to the Hazard Report.

7.21 FIRE PROTECTION

A fire in the ACOP would be controlled by the fire detection and suppression provisions of the integrated ISS EXPRESS Rack. Also, the ACOP electrical power distribution circuitry is being designed to meet the ISS interface protection requirements of SSP 57000 and the payload circuit protection requirements of NASA Letter #TA-92-038. Verification will be by assessment and review of electrical drawings. This is addressed on Hazard Report #AMS-02-10 in Appendix B.

7.22 INABILITY TO COMPLETELY INSTALL/REMOVE THE AMS-02 IN/FROM THE ACTIVE PAYLOAD ATTACH SYSTEM (PAS)

An active PAS failure could result in a partially installed AMS-02 on the ISS truss. Failure of the Capture Latch Assembly (CLA) mechanism and failure of the EVA override capability of the CLA on the active PAS would prevent removal of the partially installed AMS-02. Subsequent inadvertent release of the AMS-02 could cause it to collide with the ISS. The AMS-02 design will employ an EVA unloadable and removable capture bar or passive PAS assembly to provide one additional release mechanism. This will be verified by review of drawings, certification of flight hardware conformance to drawings and operational tests. This is addressed on Hazard Report #AMS-02-11 in Appendix B.

7.23 OTHER

The AMS-02 does not contain any pyrotechnic devices or RF transmitters. It does not have any deployable/extendible components. The AMS-02 payload is using the Space Shuttle Program (SSP) provided and ISS provided hardware for the same purposes and in the same way that it was designed.

APPENDIX A

FLIGHT PAYLOAD STANDARDIZED HAZARD CONTROL REPORT

FOR THE AMS-02

(Please find this appendix as a separate submittal)

APPENDIX B

AMS-02 UNIQUE FLIGHT PAYLOAD HAZARD REPORTS

(Please find this appendix as a separate submittal)